GATES LEARJET CORPORATION

maintenance manual

CHAPTER

27

FLIGHT CONTROLS

Record of Temporary Revisions

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FLIGHT CONTROLS - DESCRIPTION AND OPERATION

1. DESCRIPTION

NOTE: ° Due to the possible effects on stall characteristic, whenever maintenance as noted in items (1) through (4) below is performed, the aircraft must be flight tested to verify aerodynamic acceptability and stall speeds. This flight test must be conducted by a pilot approved by Gates Learjet Corporation for stall test flights. The stall warning system must be checked in accordance with the flight or ground calibration procedures in 27-31-00 of this maintenance manual when maintenance as noted in items (1) through (5) is performed.

(1) When maintenance requiring removal and installation, repair, or installation of a new wing leading edge is performed. This includes loosening and retightening any of the leading edge screws.

(2) When a tip tank strake is loosened or removed from the tip tank or a new strake is installed. Applies to 25-206 and Subsequent prior aircraft equipped with AAK 76-4, "Reduced Approach Speed System."

(3) When a new tip tank is installed. Applies to 25-206 and Subsequent and prior aircraft equipped with AAK 76-4, "Reduced Approach Speed System."

(4) When maintenance requiring repair to, or installation of, new stall strips is performed. Applies to 25-290 and Subsequent and prior aircraft equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers."

(5) When a stall warning transducer, computer, or indicator is repaired, replaced, or calibrated.

° On Aircraft 25-206 and Subsequent and prior Aircraft equipped with AAK 76-4, "Reduced Approach Speed System," replacement of the stall transducer vane is permitted in the event of a failed heater. Physical damage to the vane requires total transducer replacement because of possible damage to the transducer shaft, bearing, and potentiometer. Vane replacement requires functional test to verify that shaker and pusher actuation occurs within the prescribed tolerance of the vane plate marks.

° If angle-of-attack indicator replacement does not require an IND or R_x potentiometer adjustment and a quick check shows indicator to be at intersection of yellow/green for shaker and at intersection of red/yellow for pusher then the aircraft does not require flight test/adjustment or ground calibration adjustment.

° Perform functional test of stall warning system in accordance with the current inspection interval specified in Chapter 5.

NOTE: ° A flashing stall warning annunciator indicates proper system operation. A steady stall warning annunciator indicates system malfunction unless stall switches are off or angle of attack needle is in red area.

° Angle-of-attack vanes must be moved slowly to give correct indication. The rate of vane movement is used in stall computation. Failure to move vanes slowly will cause angle-of-attack indicator needle movement to be jumpy.

° Components of stall warning system that do not meet functional test requirements shall be replaced or recalibrated.

A. General maintenance practices of the flight control system consist of control cable temperature load correction, control cable damage limits, control surface balance check, and control surface balancing.
B. Temperature-load correction is required when checking cable tension. All control cable rigging should be performed in a hangar whenever possible due to temperature/load variations.
C. Control cable damage limits are necessary to determine when cables should be replaced. Failure of cables is rarely abrupt but is progressive over periods of extended use.
D. Control surface balance check is mandatory whenever a new control surface is to be installed, has undergone major repairs, or has been painted.
E. Control surface balancing is mandatory whenever a control surface does not meet balance limits specified by the control surface balance check.
F. Six control cables (elevator, aileron, and rudder) pass through frame 15 and are pressure sealed with a rubber seal held in place by a two-piece phenolic block. The rubber seal is designed to eliminate the need for control cable removal when replacing seals. If cabin pressure leakage becomes excessive or noise develops from air escaping around the cables, the seals should be replaced.
G. When replacement aileron, elevator, rudder, or flap cables are installed, the control cables must be pre-stretched and seated-in. To accomplish this, install control cables, adjust tension to 200 (±10) pounds, cycle system 25 times, and reduce tension to approximately 75 pounds. Proceed with control system rigging.
H. Flight controls consist of control surfaces, warning systems, and related mechanical and electrical systems that control the aircraft during flight.

I. Control Surfaces (See figure 1.)

(1) The control surfaces and their locations are as follows:

Control Surface	Location
Ailerons	Outboard trailing edge of each wing.
Elevators	Trailing edge of horizontal stabilizer.
Rudder	Trailing edge of vertical stabilizer.
Horizontal Stabilizer	Top of and perpendicular to vertical stabilizer.
Flaps	Inboard trailing edge of each wing.
Spoilers	Upper wing surface forward of flaps.

(2) Ailerons provide lateral control of the aircraft. Aileron control is provided by dual control wheels. An aileron trim tab is installed on the left aileron to further increase lateral stability. Balance tabs are installed on both ailerons to decrease force required at the control wheel.

(3) Elevators provide longitudinal control of the aircraft. Elevator is by fore and aft movement of the control columns.

(4) Rudder provides directional control of the aircraft. Rudder control is manual through the rudder pedals. Directional trim is provided by a trim tab located on the lower rudder trailing edge.

(5) Horizontal stabilizer provides pitch trim control. Horizontal stabilizer control is through an electrical actuator.

(6) Flaps provide increased wing left when partially extended during takeoff or landing approach. The increased lift provided by the extended flaps allows a lower airspeed when landing. Flaps are electrically controlled by a flap switch on the center pedestal and hydraulically operated.

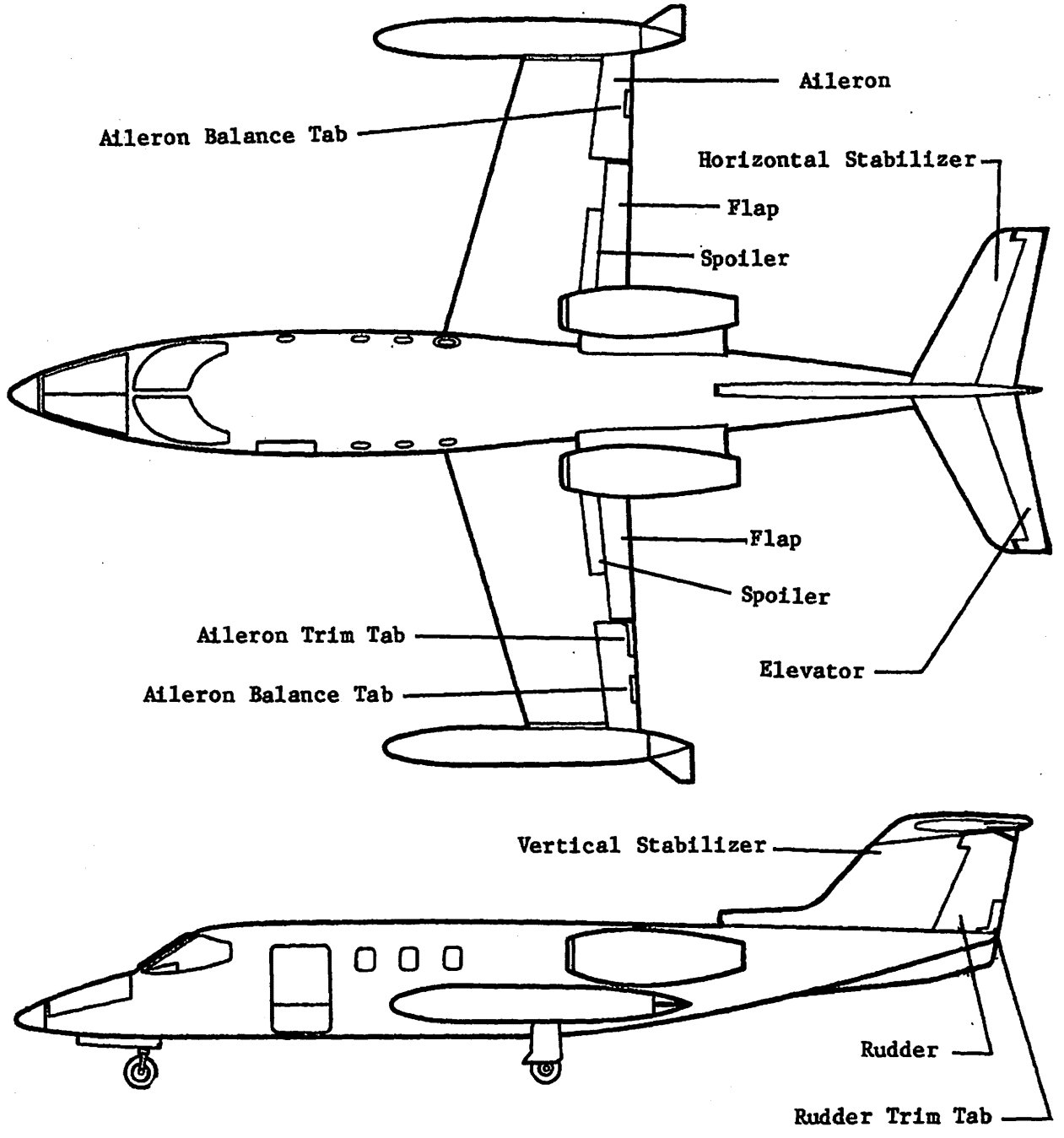
(7) Spoilers provide fast, precise speed control. Spoilers are electrically controlled by a switch on the center pedestal and hydraulically operated.

J. Trim Control Surfaces

(1) The aileron trim tab further increases lateral stability of the aileron. Aileron trim is obtained through Trim and Trim Arming Switch on either control wheel. An aileron trim indicator is located on the cockpit center pedestal.

(2) Rudder trim tab provides yaw trim for the airplane. Rudder trim is obtained through the Yaw Trim Switch on the cockpit center pedestal. A rudder trim indicator is located on the cockpit center pedestal.

(3) The movable horizontal stabilizer serves as a pitch trim controller. Pitch trim is obtained through the Trim and Trim Arming Switch (NOSE UP or NOSE DN) on the control wheel. Secondary pitch trim is controlled through a pitch trim selector (EMERGENCY) on the cockpit center pedestal. A stabilizer trim indicator is located on the cockpit center pedestal. The horizontal stabilizer is also used for autopilot pitch trim control. (Refer to Chapter 22.)



Flight Control Surfaces
Figure 1

FLIGHT CONTROLS - MAINTENANCE PRACTICES

1. Removal/Installation

A. Control Cable Pressure Seal Replacement (See figure 201)

(1) Remove carpet to gain access to aft center floorboard.

(2) Remove aft center floorboard and the forward keelbeam access cover.

(3) Remove attaching parts and phenolic block from frame 15.

(4) Remove phenolic block and rubber seal.

(5) Separate new rubber seal at three slices and position on frame 15 over control cables. Assure slices in seal have mated properly after installation.

(6) Install phenolic block halves and secure with attaching parts.

NOTE: Do not apply sealant to pressure seal.

(7) Install floorboards and carpet. Install forward keelbeam access cover.

2. Adjustment/Test

A. Temperature-Load Correction (See figure 202)

(1) All flight control rigging procedures should be performed in a hangar, when possible, because of temperature/load variation. Figure 202 provides an effective interpolation of temperature coefficients to cable tension.

(2) Temperatures shown in figure 202 are for ambient air, not airframe compartments.

(3) If aircraft has been flown at altitude, allow 40 to 60 minutes for stabilization of airframe temperatures before rigging cable tensions.

(4) Rigging may be accomplished with aircraft on jacks without affecting cable tensions.

(5) Any approved loading configuration will not affect cable tension rigging.

B. Tensiometer Adapter Calibration (See figure 203)

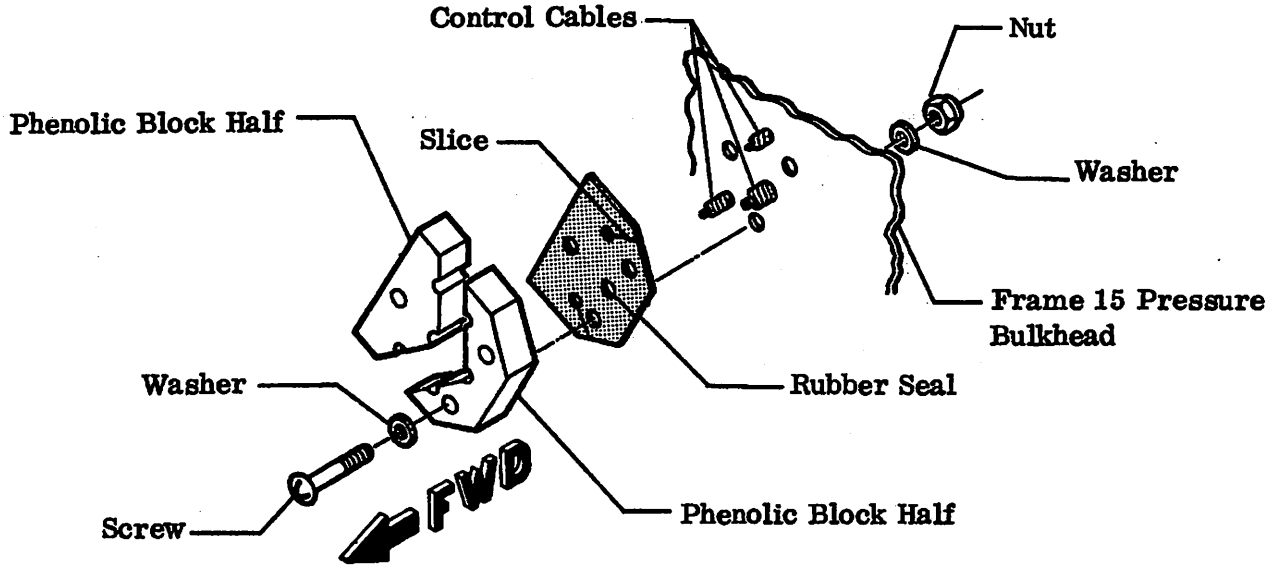
NOTE: The tensiometer adapter is intended for use on servo cables in areas that are inaccessible to normal tensiometer usage.

(1) Set up a 1/16 inch diameter cable to a 25 pound (11 Kg) tension.

(2) Install adapter on a torque wrench of no more than 30 inch-pounds (3.38 N·m) range.

(3) Install tensiometer adapter on cable with one grooved arm of adapter on each side of cable.

NOTE: The smallest torque range wrench available will allow the easiest to read and most accurate values.



TYPICAL TWO PLACES

Control Cable Pressure Seal (Frame 15 Feed-Through)
Figure 201

TEMPERATURE-LOAD CORRECTION TABLE									
CABLE SIZE	POUNDS								
3/32	-10	-7.3	-4.5	-1.5	0	+1.5	+4.5	+7.3	+10
1/16	-5	-3.5	-2.1	-0.75	0	+0.75	+2.1	+3.5	+5
1/8	-17.5	-12.6	-7.5	-2.5	0	+2.5	+7.5	+12.6	+17.5
3/16	-37.7	-27	-16	-5.5	0	+5.5	+16	+27	+37.5
TEMPERATURE (°F)	40°	50°	60°	70°	75°	80°	90°	100°	110°

EXAMPLE:

AILERON SYSTEM CABLE SIZE	1/8 inch
AMBIENT TEMPERATURE	98°
CABLE TENSION (From Aileron Rigging Procedure)	75 ±10 pounds
TEMPERATURE LOAD CORRECTION (Interpolation from Table)	+11.6 pounds

100° = +12.6
90° = + 7.5
5.1 Difference
98° = .8 Greater than 90°
4.08 = .8 of 5.1
7.5 + 4.08 = 11.58 (11.6)

FINAL RIGGING TENSION 75 + 11.6 = 86.6 ±10 pounds.

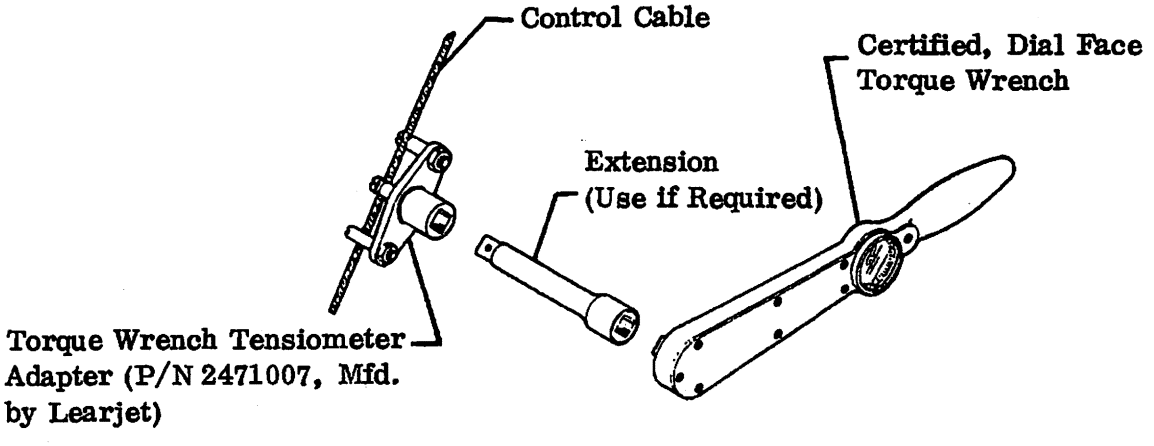
Temperature-Load Correction
Figure 202

(4) Rotate adapter until third arm (ungrooved) contacts cable. Record torque value change in third column of calibration chart in figure 203. Perform procedure three times.

NOTE: It is important that grooved arms are properly aligned with cable, and that contacting torque is noted or incorrect calibration will result.

Torque Value Change* (in inch-pounds)	-3	**	+2
Cable Tension	15	25	35

*Figures are valid for 1/16 inch diameter cable only.
**Torque value arrived at in step 2.B.(4).



Tensiometer Adapter
Figure 203

C. Control Surface Balancing (See figure 204)

(1) Remove applicable control surface from aircraft.
(2) Replace all hinge bolts at control surface hinge points.
(3) Position trim tab to neutral and assure that all attaching hardware is installed.
(4) Tape balance tab to neutral and tape balance tab pushrods to surface.
(5) Remove aileron yokes.
(6) Position control surface on balance stand with hinge bolts resting on knife edges.

NOTE: Place aileron and elevator on stand with right side up. On rudder control surfaces, use upper hinge point bolt and bolt through the rudder torque tube as balance points.

(7) Move a known weight along surface until control surface is balanced to level condition.
(8) Measure from centerline of hinge point to center of gravity of weight.
(9) Multiply this distance in inches by the weight in pounds to obtain inch-pounds overbalance.
(10) If the overbalance is not within the prescribed limits shown in figure 204, correct condition by adding to or subtracting from the balance weights in the leading edge.

NOTE: When balancing rudder, weight adjustments are made to uppermost weight. (Refer to 27-20-01.)

3. Inspection/Check

A. Control Surface Balance Check

(1) The flight control surfaces are designed with an overbalance (nose heavy condition). These surfaces are balanced after initial paint application. The amount of overbalance and unit weight is recorded on the outboard rib surface of the ailerons and elevators and on the bottom rib of the rudder.
(2) When any portion of a control surface is repainted, repaired, or replaced, a check must be made to determine if the balance is still within prescribed limits. Balancing must be accomplished when a surface area of 12 square inches or greater is repainted.
(3) Check control surface balance as follows:
(a) Accurately weigh each part added or removed to within ±0.03 pound.
(b) Locate center of gravity of net weight change and measure distance to hinge line.
(c) Determine the net moment change by using formula M = WR.
1) M = Change in overbalance
2) W = Net weight change
3) R = Distance between hinge line and center of gravity.

NOTE: "R" is negative when center of gravity of change falls aft of the hinge line.

(d) Add or subtract "M," according to its sign, from the overbalance; and change the balance weight, if required, to stay within the prescribed overbalance limits.

(e) Remove the old overbalance and unit weight figures recorded on the control surface and apply the new overbalance and unit weight figures.

B. Control Cable System Inspection

(1) Aircraft cable systems are subject to a variety of environmental conditions and forms of deterioration thay may be easy to recognize as wire/strand breakage or the not-so-readily visible type of wear, corrosion, and/or distortion.

(2) Critical areas of wire breakage are those sections of the cable which pass through fairleads and around pulleys. Examine cable for broken wires by passing a cloth along the length of the cable. This will clean the cable for a visual inspection, and detect broken wires when the cloth snags on the broken wire. If snags are found, closely examine the cable to determine the full extent of the damage. (Refer to Cable Damage Limits, this section).

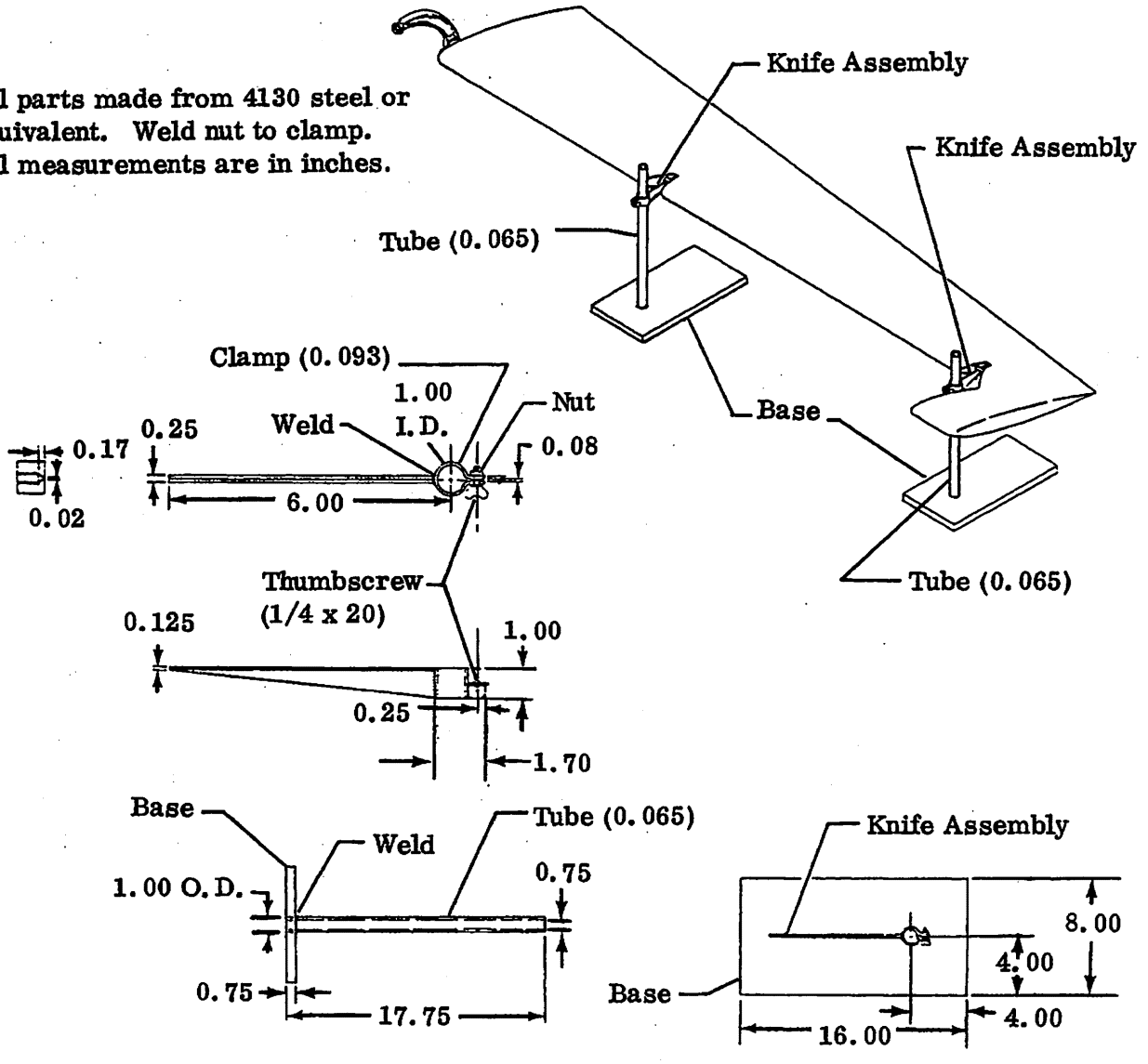
(3) The absence of snags is not positive evidence that broken wires do not exist. Wire breakage in this case is not apparent unless the cable is removed and bent into a loop, thus forcing the broken wires to protrude from the cable.

(a) External Wear Patterns - Wear will normally extend along the cable equal to the distance the cable moves at the location and may occur on one side of the cable only or on its entire circumference. Replacement of cables is recommended when the individual wires in each strand appear to blend together.

(b) Internal Cable Wear - As wear it taking place on the exterior surface of a cable, the same condition is taking place internally, particularly in the section of the cable which passes over pulleys and quadrants. This condition is not easily detected unless the strands of the cable are separated. This wear is a result of the relative motion between the inner wire surfaces and in some instances the wear can be greater than that occurring on the surface of the cable.

(c) Corrosion - Any cable that has broken wire in a section not in contact with pulleys or fairleads should be carefully inspected for corrosion. It may be necessary to remove and bend the cable to properly inspect it for internal strand corrosion as this condition is usually not evident on surface of cable. Replacement of cables is recommended if evidence of rust or corrosion is found.

All parts made from 4130 steel or equivalent. Weld nut to clamp.
All measurements are in inches.



CONTROL SURFACE	BALANCE LIMITS (INCH-POUNDS)
Aileron	15 to 16 (Overbalance)
Elevator	10.0 to 10.5 (Overbalance)
Rudder	2 to 2.5 (Overbalance)
Aileron Tab	.0 to .1 (Overbalance)
Rudder Tab	-0.5 to +0.5 (Static Balance)

Control Surface Balancing
Figure 204

(4) Inspect pulleys for roughness, sharp edges, and presence of foreign material embedded in the grooves. Examine pulley bearings to assure proper lubrication, smooth rotation, free of flat spots, dirt and paint spray. Pulleys which turn through a small arc should periodically be rotated to provide a new bearing surface for the cable. Maintain pulley alignment to prevent cable from riding on pulley flanges and rubbing against cable guards or adjacent structure.

(a) Pulley wear patterns - Various cable system malfunctions may be detected by analyzing pulley conditions. These include such discrepancies as too much tension, misalignment of pulley bearing and size mismatch. Examples of these conditions are shown in figure 206.

C. Control Cable Damage Limits

(1) Failure of control cables may be progressive over periods of extended use. Some broken wires may be evident soon after placing the cables in service due to some wires being under greater tension or due to variation in wire hardness. During normal service, a few additional broken wires will be encountered after these overstressed or overhard wires have broken.

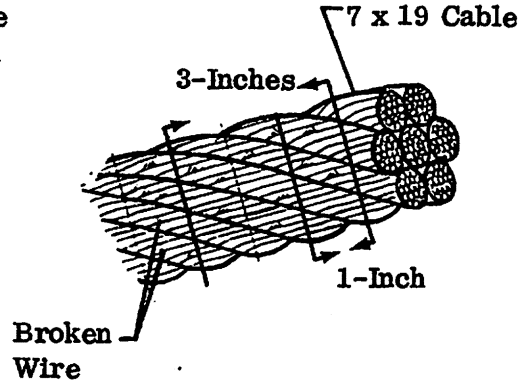
(2) Control cable strength is as much as 50% in excess of design loads, so control cables may have broken wires and still be capable of carrying the design load of the cable.

(3) The control cables vary from 1/16 to 3/16 inch in diameter. Cables of 1/8 inch and larger are composed of 7 strands of 19 wires each totaling 133 wires. Cables of 1/16 and 3/32 inch in diameter are composed of 7 strands of 7 wires each totaling 49 wires.

(4) In accordance with current inspection intervals, check for frayed cables. Cables may be inspected for broken wires by passing a cloth along the length of cable. Broken wires will be indicated where the cloth is snagged.

NOTE: Replace any 7 x 19 cable that shows more than 6 wires broken in any 1-inch length.

Replace any 7 x 7 cable that shows more than 3 wires broken in any 1-inch length.



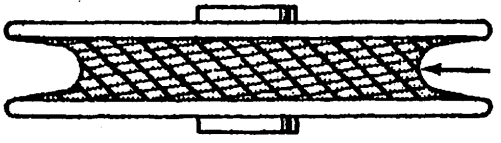
Control Cable Damage Limits
Figure 205

(5) Assure that a close inspection is made in the length of cable normally passing over a pulley or through a fairlead.

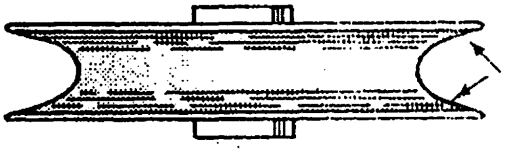
(6) Replace an 7x19 cable that shows more than 6 wires broken in any 1-inch length of cable.

(7) Replace any 7x7 cable that shows more than 3 wires broken in any 1-inch length of cable.

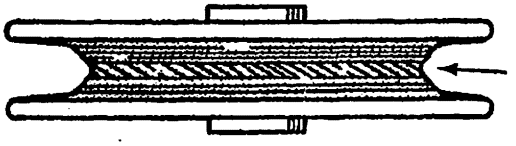
NOTE: When replacement aileron, elevator, rudder or flap cables are installed, the control cables must be pre-stretched and seated-in. To accomplish this, install control cables, adjust tension to 200 (±10) pounds, cycle system 25 times, and reduce tension to approximately 75 pounds. Proceed with control system rigging.



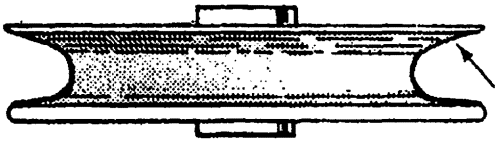
Excessive Cable Tension



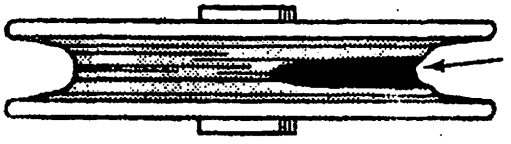
Pulley Misalignment



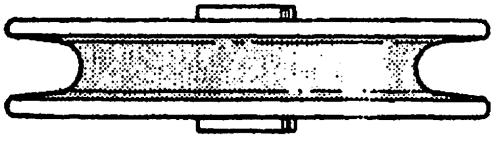
Pulley Too Large for Cable



Cable Misalignment



Frozen Bearing



Normal Condition

Pulley Wear Patterns
Figure 206

D. Trim Systems Operational Check (Aircraft 25-337, 25-342 and Subsequent and Prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement.)

NOTE: On Aircraft 25-061 and 25-070 thru 25-341 not modified for Horizontal Stabilizer Trim and Autopilot Improvement, refer to 27-40-00 for Horizontal Stabilizer Pitch Trim Operational Check.

(1) Set Pitch Trim Selector Switch, labeled P. TRIM-NORM/OFF/EMER, to EMER.

(2) Set EMERGENCY Pitch Trim Switch (pedestal) to NOSE UP, then NOSE DOWN. Check for horizontal stabilizer movement in both directions. Movement shall be accompanied by the audio clicker.

(3) Set EMERGENCY Pitch Trim Switch (pedestal) to NOSE UP. While trimming, depress Control Wheel Master Switch; trim motion shall stop when the Control Wheel Master Switch is depressed. Repeat procedures for NOSE DOWN condition; trim motion shall stop.

(4) Set Pilot's and Copilot's Control Wheel Trim Switches (with arming button depressed) to NOSE UP and then NOSE DOWN. Trim motion shall not occur.

(5) Set Pitch Trim Selector Switch to OFF.

(6) Actuate the Control Wheel Trim and Trim Arming Switches; trim motion shall not occur.

(7) Actuate the EMERGENCY Pitch Trim Switch. Trim motion shall not occur.

(8) Set Pitch Trim Selector Switch to NORM.

(9) Set EMERGENCY Pitch Trim Switch to NOSE UP and then NOSE DOWN position. Trim motion shall not occur.

(10) Without depressing the arming pushbutton, move Pilot's Control Wheel Trim Switch to LWD, RWD, NOSE UP, and NOSE DOWN. Trim motion shall not occur.

(11) Depress Trim Arming pushbutton; trim motion shall not occur.

(12) Set the Pilot's Control Wheel Trim and Trim Arming Switches to NOSE UP and then to NOSE DOWN. Trim motion shall occur and audio clicker shall sound.

(13) Actuate the Pilot's Control Wheel Trim and Trim Arming Switches. While trimming, depress either Control Wheel Master Switch; trim motion shall stop when Control Wheel Master Switch is depressed.

(14) Actuate the Pilot's Control Wheel Trim and Arming Switch for a LWD and then RWD condition. Roll trim motion shall occur without the audio clicker.

(15) Repeat steps (10), (11), (12), (13), and (14) using Copilot's Control Wheel Trim Switch.

(16) Trim by positioning Copilot's Control Wheel Trim Switch, then trim in opposite direction using the Pilot's Control Wheel Trim Switch. Pilot's trim shall override copilot's trim. Repeat for each trim position.

(17) Set flaps to 8°.

(18) While trimming NOSE UP or NOSE DOWN with either Control Wheel Trim Switch, hold the PITCH TRIM OVSP TEST Switch (forward pedestal) to TEST. The amber PITCH TRIM OVSP light shall illuminate. Verify that pitch trim needle moves more rapidly with flaps extended. Release PITCH TRIM OVSP TEST Switch.

(19) Run the pitch trim toward the full nose down and nose up travel to verify operation of the T.O. out-of-trim light on the annunciator panel.

(20) Operate each half of the YAW TRIM Switch separately to NOSE LEFT and NOSE RIGHT. Trim motion shall not occur.

(21) Operate both halves of the YAW TRIM Switch simultaneously to NOSE LEFT and NOSE RIGHT. Trim motion shall occur.

(22) Assure that all axes are set for takeoff. Check that T.O. light is extinguished.

NOTE:
- If the airplane is loaded toward the aft CG, the Pitch Trim Indicator should be placed near the forward position of the Green T.O. Band.
- If the airplane is loaded toward the forward CG, the Pitch Trim Indicator should be placed near the aft portion of the Green T.O. Band.

AILERON AND TAB CONTROL SYSTEM - DESCRIPTION AND OPERATION

1. Description (See figure 1)
 - A. The aileron system provides lateral control of the aircraft through three subsystems: an aileron control system, an aileron trim tab control system, and an aileron balance tab control system.
 - B. Aileron control is provided manually through the control wheels or automatically through the autopilot system.
 - C. Aileron trim tab control is provided through the trim and trim arming switch on each control wheel.
 - D. Aileron balance tab control is provided through push-pull tubes attached to the fixed wing surface.
 - E. An aileron-rudder spring interconnect is installed to increase free aileron response to rudder input. (Refer to 27-11-00.)

2. Component Description
 - A. The ailerons are hinged at three places on the aft spar of each wing outboard of the flaps and are balanced to a nose-heavy overbalance by lead weights in the leading edge. Separate closed-loop cables connect both aileron drive pulleys to a common sector on the aircraft centerline. A followup connected to the left aileron drive pulley provides output for the autopilot roll computer. A trim tab is located on the inboard trailing edge of the left aileron, and a balance tab is located on the trailing edge of each aileron. Three small drain holes are drilled in the lower surface of each aileron just aft of the skin splice. Additional drain holes and siphoning scupper drain fairings are installed on the lower trailing edge of each aileron. Three drain holes and fairings are installed on the LH aileron; two drain holes and fairings are installed on the RH aileron. Brush seals are installed on both ends of each aileron. On Aircraft 25-206 thru 25-289 modified per AAK 79-10 (Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers) and Aircraft 25-061, 25-070 thru 25-205 modified per AAK 76-4 (Reduced Approach Speed System) and AAK 79-10, additional brush seals are installed in the wing structure along the leading edge of the aileron. The brush seals require periodic lubrication in accordance with lubrication intervals specified in Chapter 12.
 - B. The aileron trim tab is hinged to the inboard trailing edge of the left aileron by a continuous-type hinge. The trim tab is controlled by a push-pull tube connected to a rotary-type electrical actuator located in the aileron leading edge.
 - C. The aileron balance tabs are hinged to the trailing edge of each aileron by a continuous-type hinge. The balance tabs are controlled by push-pull tubes attached to the fixed wing surface.
 - D. A U-shaped control wheel on each control column controls the ailerons. Control wheel motion is transferred to the ailerons through a closed-loop drive cable to a sector located on the fuselage centerline. Maximum control wheel movement is 110° both right and left of neutral.

3. Operation

A. Aileron Control System (See figure 1)

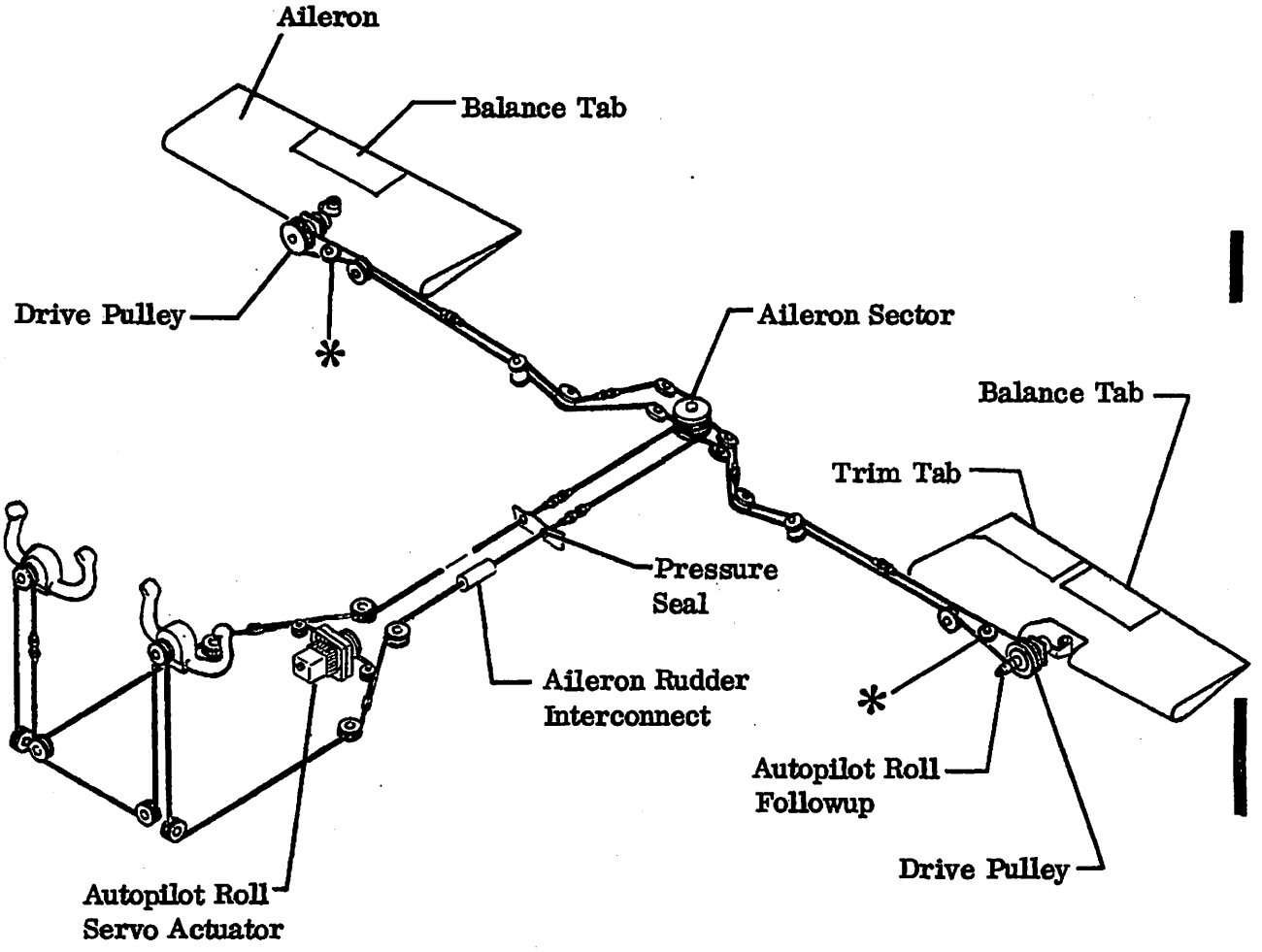
(1) When either aileron control wheel is rotated, the crossover cable rotates the other control wheel and the control cable rotates the aileron sector located on the fuselage centerline. The ailerons are interconnected through the aileron sector by separate closed-loop cables. The cables from the aileron sector are connected to the aileron drive pulleys which rotate and raise or lower the ailerons. Maximum control wheel movement is 110° both left and right of neutral. Maximum aileron movement is 18° (±1°) up and 18° (±2) down.

B. Aileron Trim Tab System (See figure 2)

(1) The trim and trim arming switch on the control wheel controls the aileron trim tab actuator in the left aileron. Depressing the trim and trim arming switch energizes the actuator to move the trim tab either up or down. A trim tab position transmitter is attached to the actuator shaft and provides input for the aileron trim tab position indicator on the pedestal. Trim tab travel is 8° (±1°) both up and down.

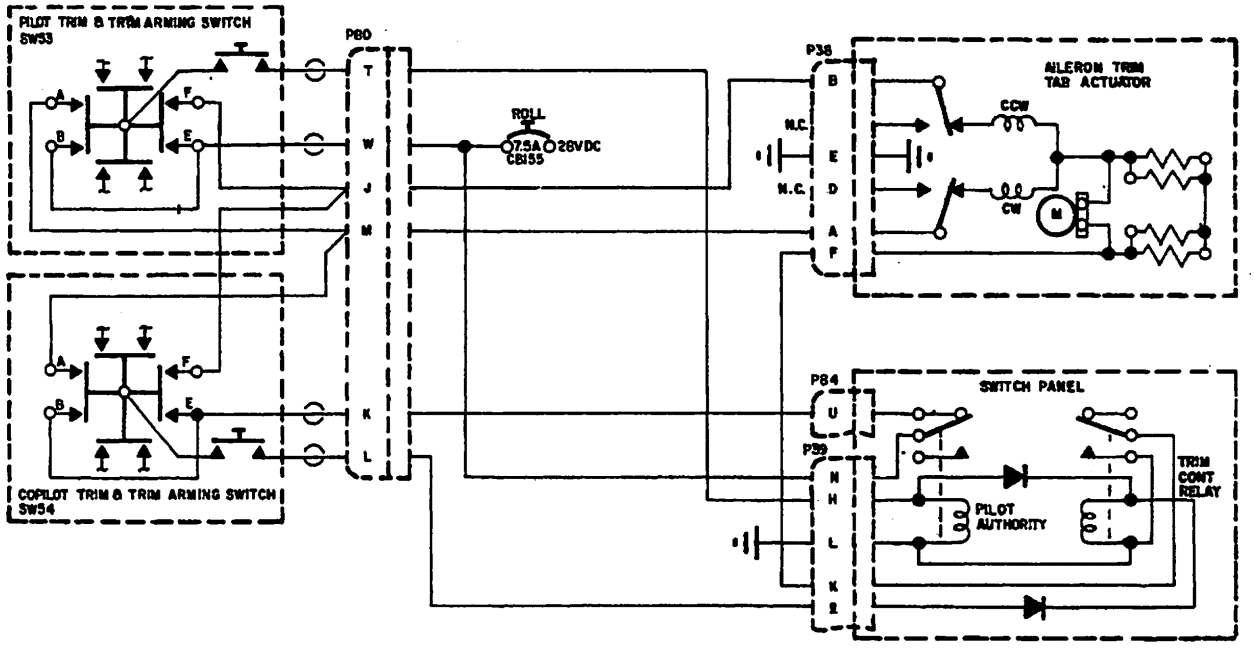
C. Aileron Balance Tab System

(1) The aileron balance tabs are hinged to the aileron and connected to the fixed wing surface by push-pull tubes. As the ailerons are moved the balance tabs move proportionately in the opposite direction. With the aileron fully up the balance tab will be 15° (±2°) down relative to the ailerons. With the aileron fully down the balance tab will be 15° (+30'/ -3°30') up relative to the ailerons.

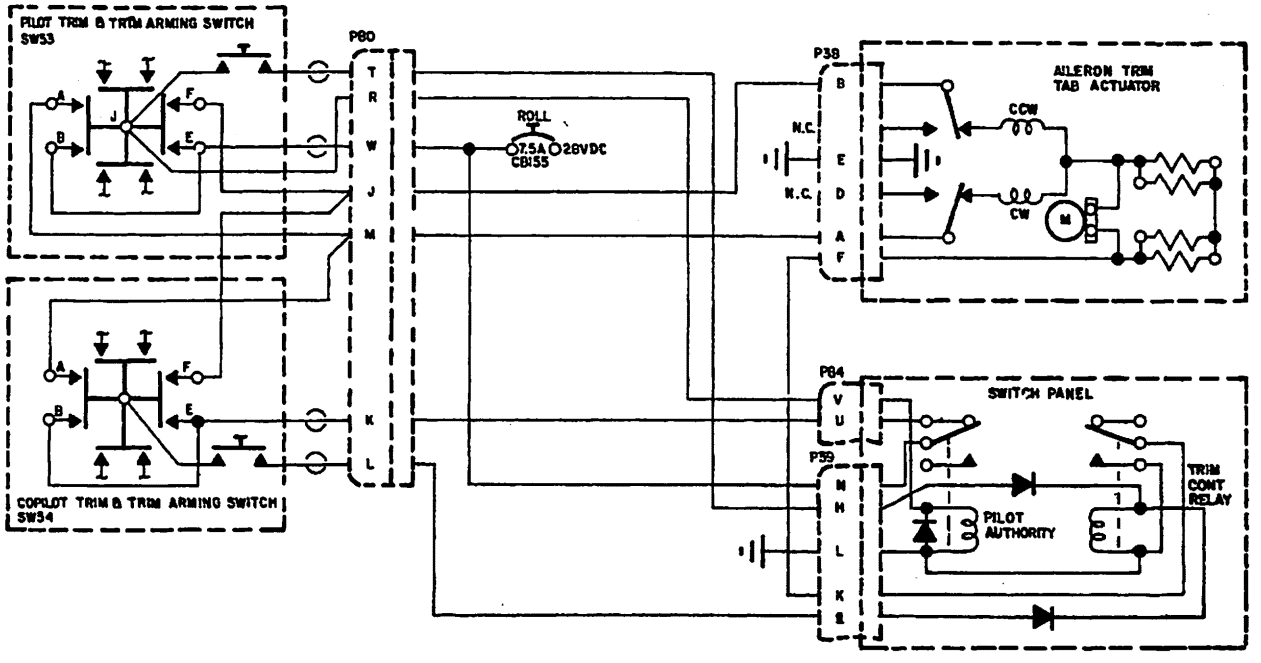


* This pulley effective on 25-225 and Subsequent and previous aircraft modified per AMK 77-1.

Aileron Control System Installation
Figure 1

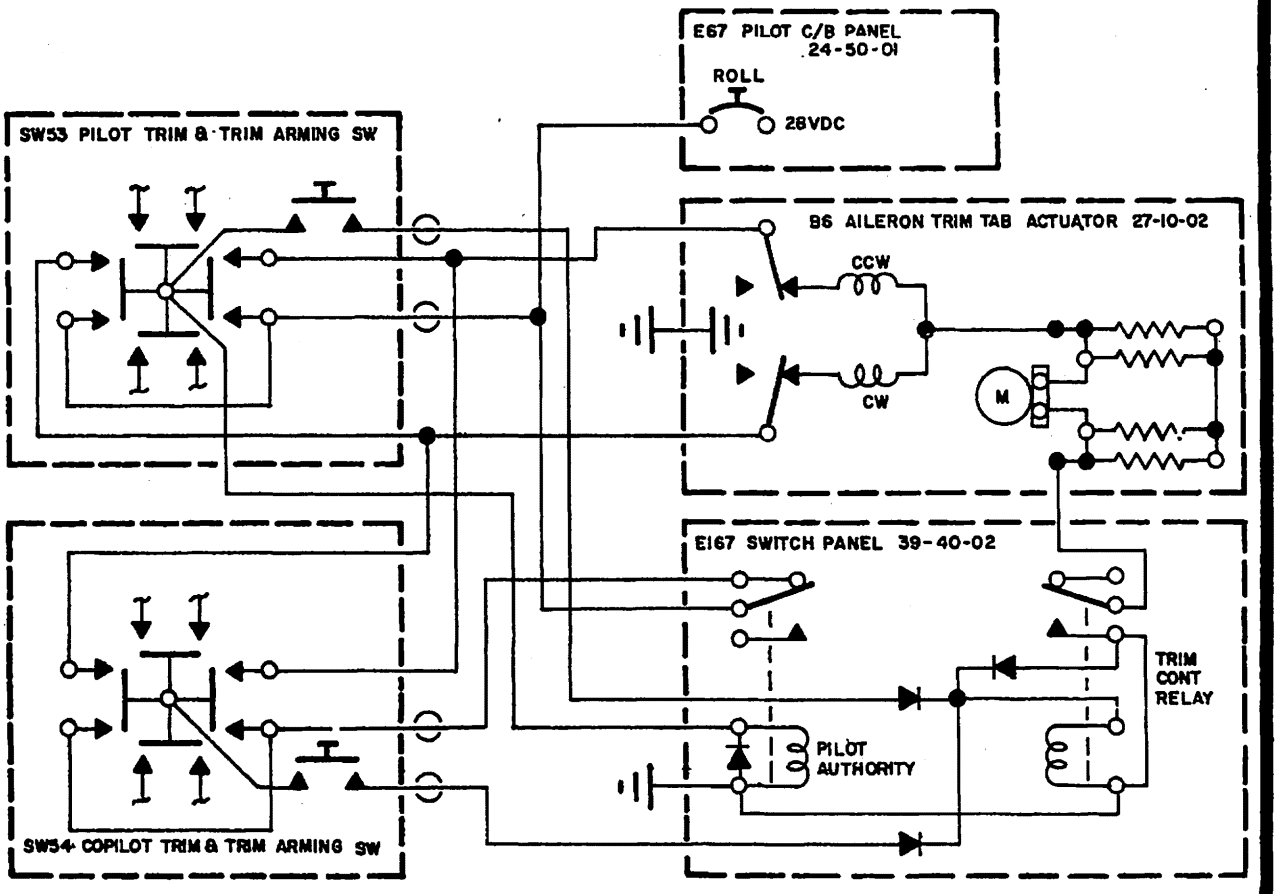


Aircraft 25-061 and 25-070 thru 25-201 NOT modified for Horizontal Stabilizer Trim and Autopilot Improvement



Aircraft 25-202 thru 25-341, except 25-337, NOT modified for Horizontal Stabilizer Trim and Autopilot Improvement

Aileron Trim Tab Electrical Control Schematic
Figure 2



Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement

Aileron Trim Tab Electrical Control Schematic
Figure 3

AILERON AND TAB CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Adjustment/Test

NOTE: ● Adjustment procedures are typical for left and right ailerons.
● Access for aileron rigging is gained by removal of access covers on copilot's control column, cabin center aisle, fuselage center section, and underside of wings.

A. Tools and Equipment required for Adjustment/Test.

NAME	NUMBER	MANUFACTURER	USE
Contour Plate	2471003-1	Gates Learjet Corp.	To locate control surface neutral.
Clinometer	2471004-1	Gates Learjet Corp.	To measure angles of control surfaces.
Tensiometer	2471007	Gates Learjet Corp.	To measure cable tension.
Spring Scale	Accurate within 1/2 pound.	Commercially Available.	To measure system friction.

B. Rig Aileron Control System (See figure 201)

(1) Remove access covers from copilot's control column, cabin center aisle, fuselage center section and underside of wings.
(2) Disconnect aileron autopilot servo cable.
(3) Center control wheels by laying a straight edge across the top of both control wheels. Clamp straightedge to control wheels in a suitable manner.
(4) Insert rigging pin in aileron sector.

NOTE: On Aircraft 25-221 and Subsequent, a hole is provided in the keel beam access cover approximately 6 inches forward of frame 22. This hole allows the rigging pin to be inserted in the aileron sector without removal of the access panel.

(5) Attach contour plate to wing.
(6) Adjust appropriate turnbuckles until both ailerons are synchronous at trailing edge. Synchronize ailerons to a droop neutral position of 0.50 (±0.06) inch below contour plate.

(7) After both ailerons have been synchronized, adjust aileron control cable tension to 75 (±10) pounds at 75° F. Adjust aileron control cable tension in wings to 75 (±10) pounds at 75° F. (Refer to 27-00-00 for cable tension temperature-load correction table.)

NOTE: When cable turnbuckles are loosened or tightened to adjust cable tension, recheck aileron synchronization. If necessary, readjust and synchronize ailerons. Repeat these procedures alternately until ailerons are synchronized and correct cable tension is obtained.

(8) Attach autopilot roll servo cable to aileron control system cables.
(9) Check capstan clutch setting. (Refer to Chapter 22.)
(10) Adjust autopilot roll servo cable tension to 25 (±5) pounds at 75° F. (Refer to 27-00-00 for cable tension temperature-load correction table.)

NOTE: When autopilot roll servo cable tensions is adjusted, primary aileron control cable forward of attach point will be 100 (±10) pounds and primary cable tension aft of attach point will be 75 (±10) pounds.

(11) Check tension on aileron control system cables aft of servo cable attachment for tension as determined in step 7.
(12) Check tension on aileron control system cables forward of servo cable attachment tensions for the sum of tensions determined in steps (7) and (10).

NOTE: The difference of cable tension noted in steps 11 and 12 is normal and is caused by the tension on the servo cable.

(13) Safety turnbuckles and remove rigging pins.
(14) Attach clinometers to ailerons to measure travel.
(15) Adjust up stops for both ailerons to obtain 18° (±1°) up travel. On Aircraft 25-198 and Subsequent and Aircraft modified per SB 23/24/25-266, safety wire stop bolts.

(16) Adjust down stops for both ailerons to obtain 18° (±2°) down travel. Down stops should be set to match up stops on opposite aileron. On Aircraft 25-198 and Subsequent and Aircraft modified per SB 23/24/25-266, safety wire stop bolts.

NOTE: Travel is measured from "droop neutral" position 0.50 (±0.06) inch below contour plate. If an aileron will achieve required travel in only one direction, it indicates that yoke is not centered on hinge line. The shims between the yoke bearings

and the center hinge casting should be changed as necessary to obtain the correct travel.

(17) Assure that fixed stops in control column heads do not contact until after primary stops on aileron drive pulley are reached.
(18) Operate ailerons several times by moving control wheel and checking for freedom of movement.
(19) Perform Aileron Control System Friction Check.
(20) Install all floorboards and access covers.

2. Inspection/Check

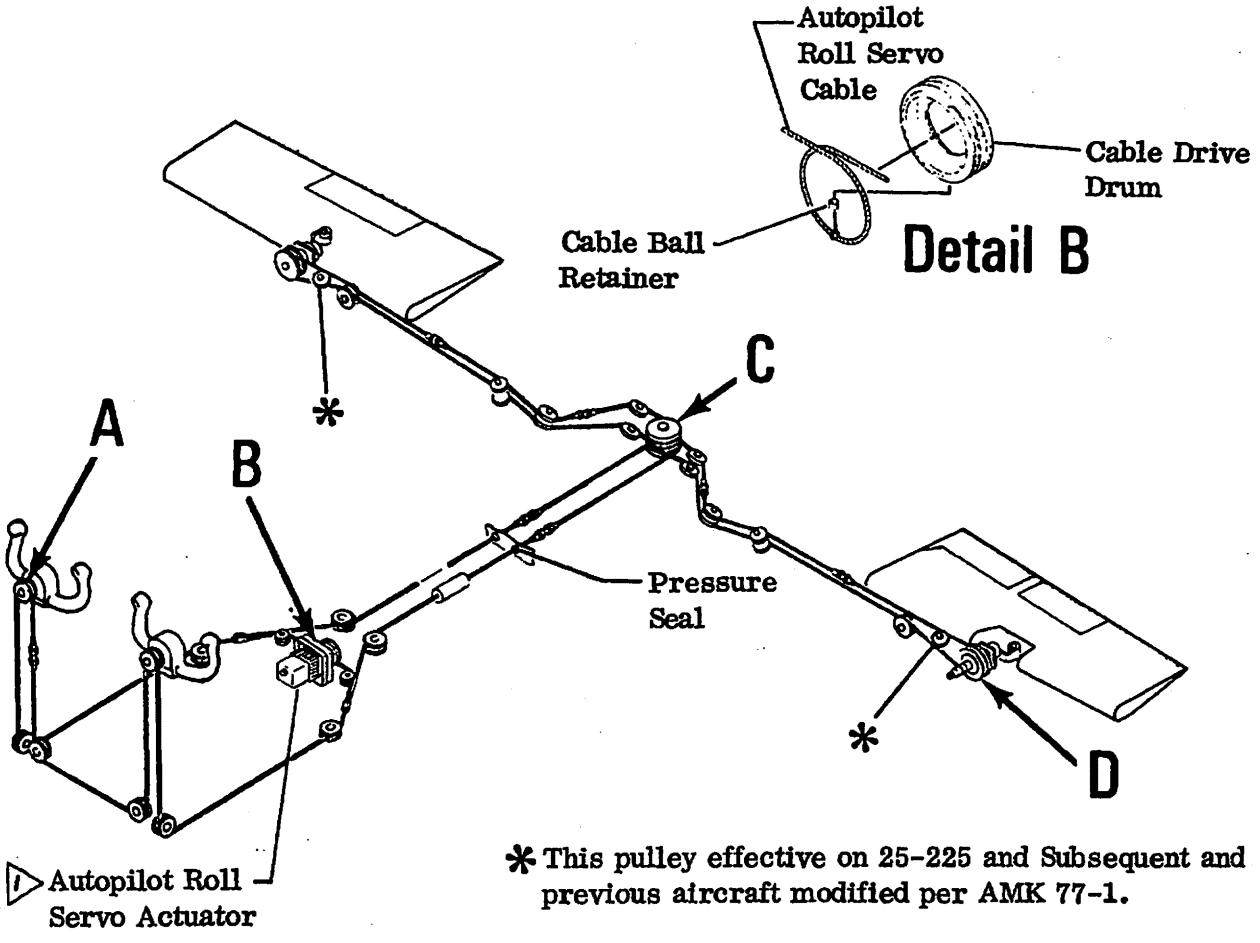
A. Aileron Control System Operational Check

(1) Turn control wheel to right; right aileron shall move up and left aileron shall move down.
(2) Turn control wheel back to neutral; ailerons shall streamline.
(3) Turn control wheel to left; left aileron shall move up and right aileron shall move down.

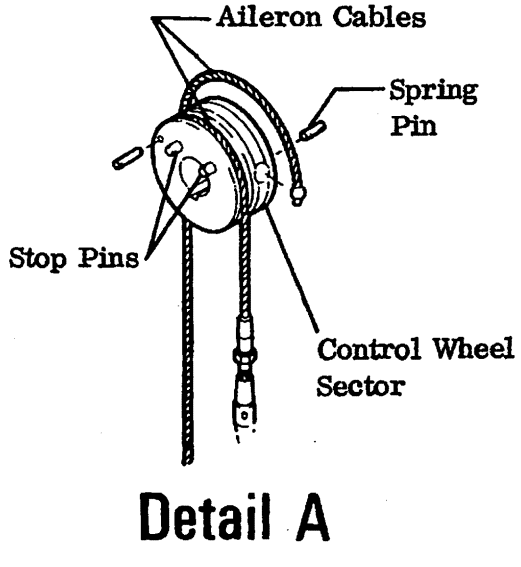
B. Aileron Control System Friction Check.

NOTE: Perform the following check with autopilot not engaged.

(1) With ailerons and rudder systems centered, place a push-pull spring scale (accurate within 1/2 pound) on horizontal arm of pilot's control wheel 4.5 inches from vertical centerline of pilot's control wheel hub.
(2) Slowly apply a downward force on spring scale.
(3) Read force applied on spring scale when control wheel starts to move. Maximum allowable force is 12 pounds.
(4) If applied force is more than 12 pounds, check for proper cable tensions; check pulleys and sector for freedom of movement and alignment; visually inspect system for binding; check system for proper lubrication; and ensure that aileron and rudder control systems are properly rigged.
(5) After all discrepancies have been corrected, perform steps (1) through (3) and ensure that applied force is less than 12 pounds.



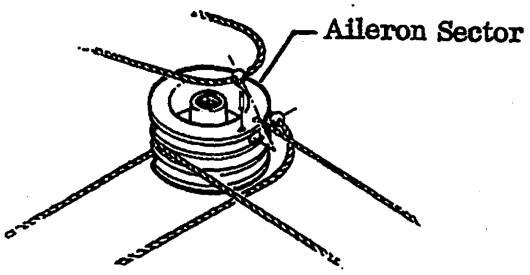
* This pulley effective on 25-225 and Subsequent and previous aircraft modified per AMK 77-1.



1 When autopilot roll servo actuator is removed, cover holes in actuator mounting bracket with tape. This will prevent foreign matter from entering capstan slip clutch gear.

Aileron Control System Rigging
Figure 201 (Sheet 1 of 6)

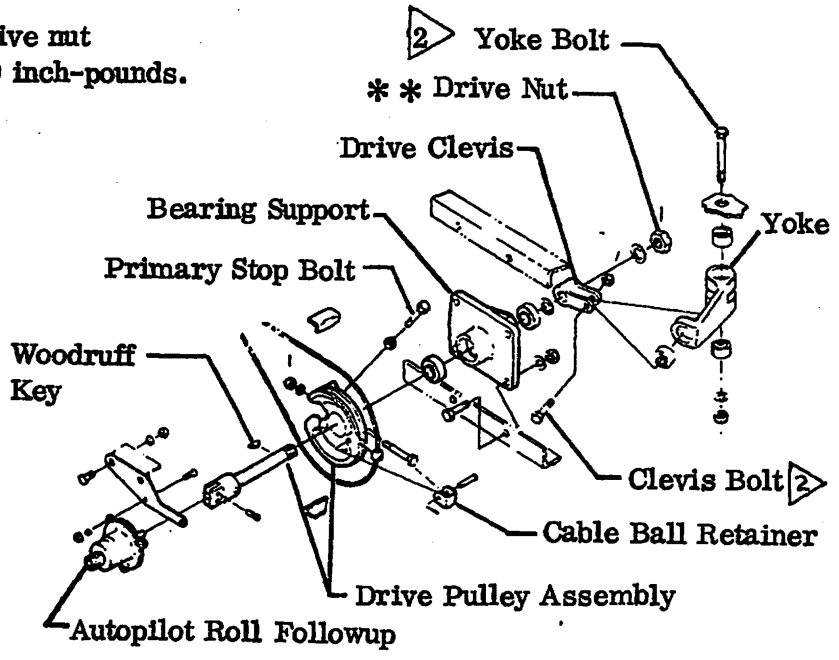
8-13B-2
A8-04B
B8-74A



Detail C

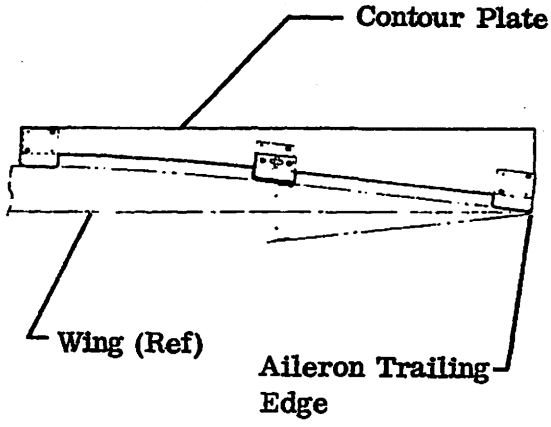
2 Use extreme care to avoid overtorquing aileron drive yoke bolt and hinge bolt or excessive control force may be required. Torque to 30-40 inch-pounds.

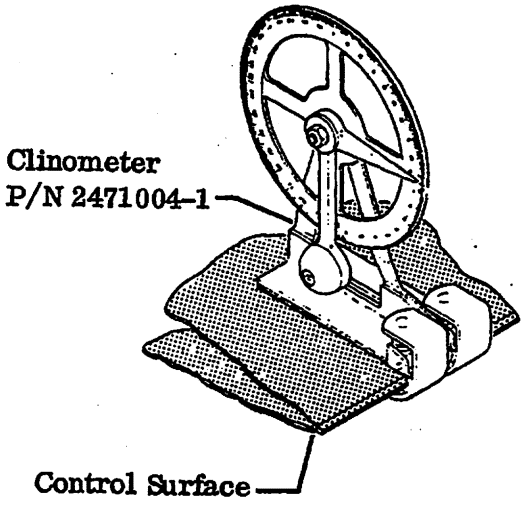
** Torque drive nut to 100-120 inch-pounds.



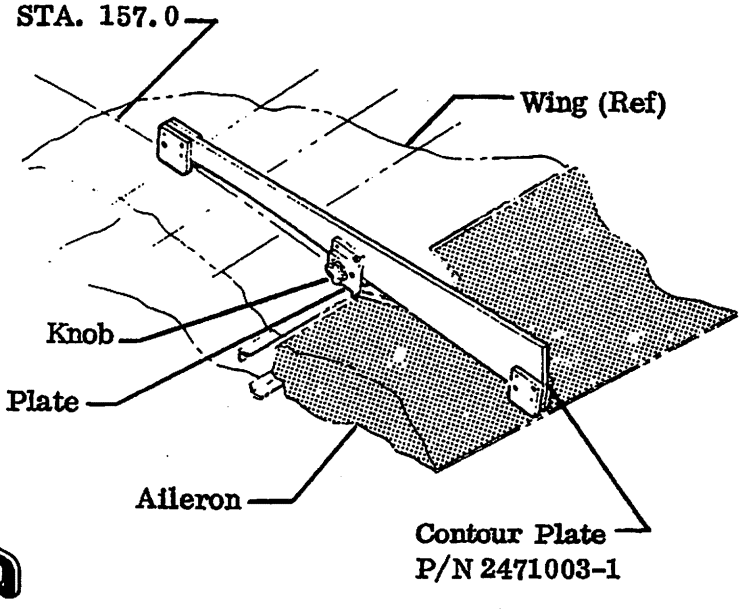
Detail D

Aileron Control System Rigging
Figure 201 (Sheet 2 of 6)



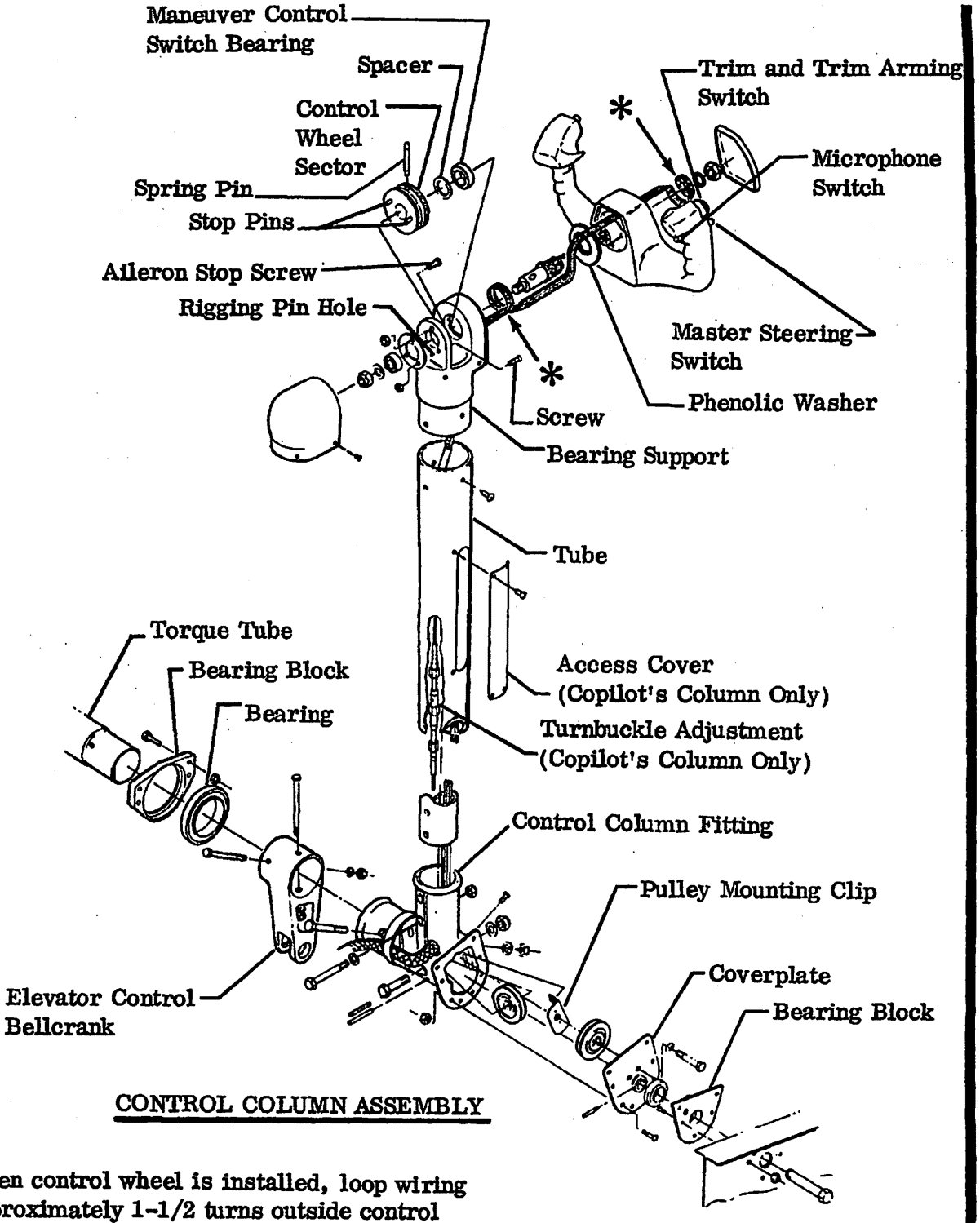


Install knob and plate on outboard side. (LH wing shown)





Aileron Control System Rigging
Figure 201 (Sheet 3 of 6)



CONTROL COLUMN ASSEMBLY

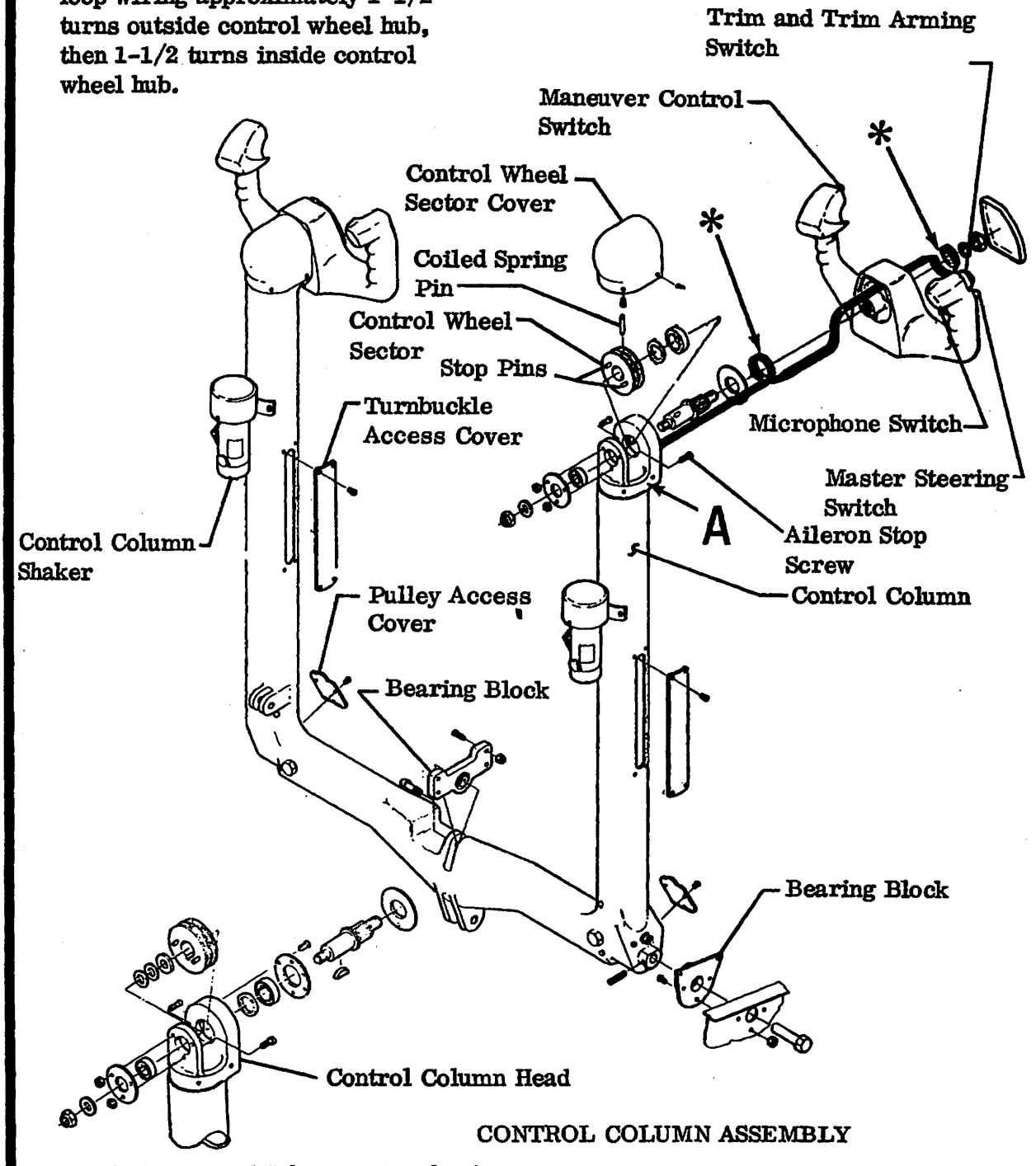
* When control wheel is installed, loop wiring approximately 1-1/2 turns outside control wheel hub, then 1-1/2 turns inside control wheel hub.

Aileron Control System Rigging
Figure 201 (Sheet 4 of 6)

8-11C
8-64B-1

EFFECTIVITY: 25-061, 25-070 thru 25-081 NOT modified per SSK 947 or AAK 76-4

*When control wheel is installed, loop wiring approximately 1-1/2 turns outside control wheel hub, then 1-1/2 turns inside control wheel hub.



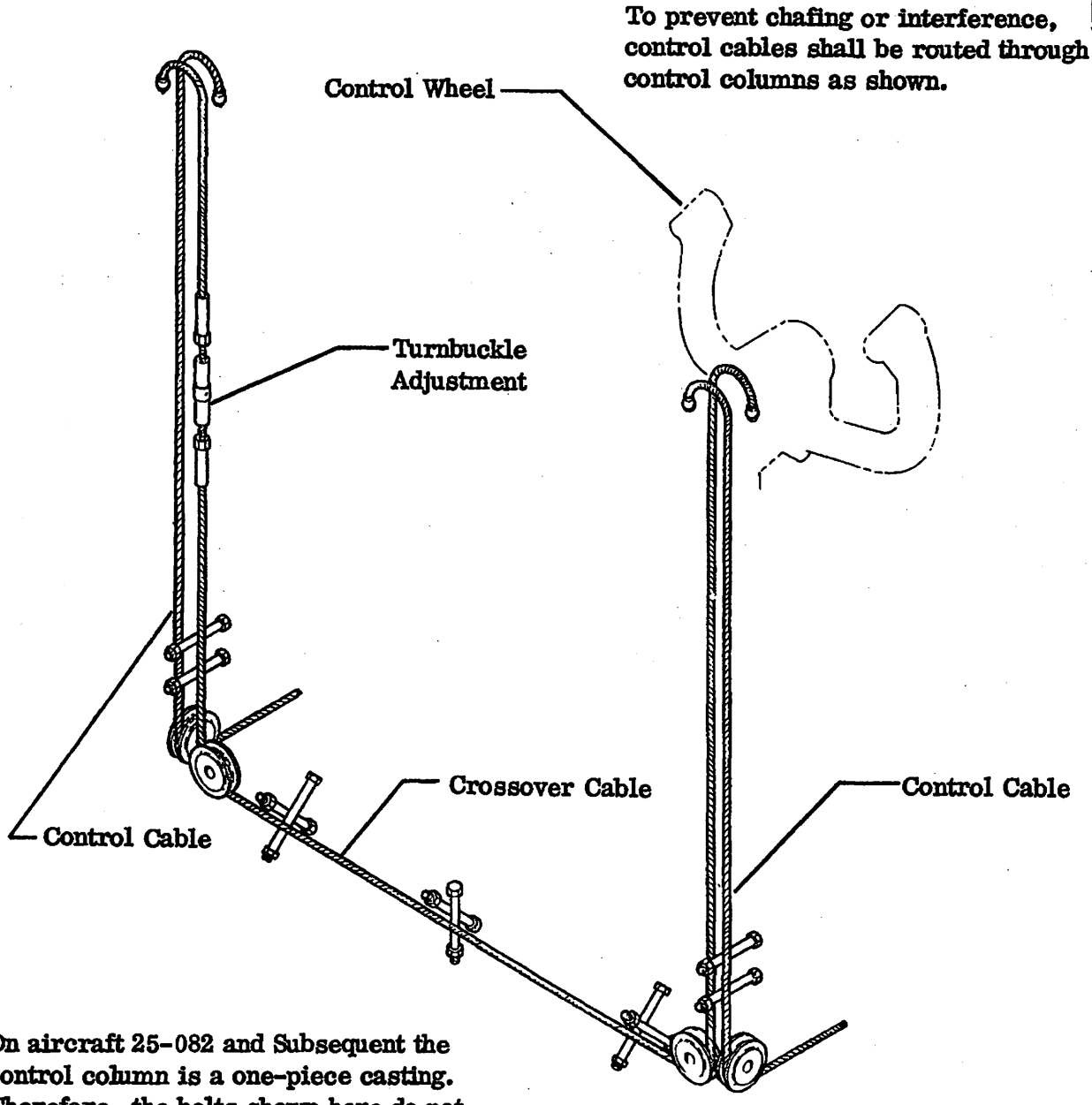
CONTROL COLUMN ASSEMBLY

Aircraft 25-187 and Subsequent and prior
Aircraft modified per SSK 947 or AAK 76-4

Detail A

Aileron Control System Rigging
Figure 201 (Sheet 5 of 6)

AM-64B



On aircraft 25-082 and Subsequent the control column is a one-piece casting. Therefore, the bolts shown here do not exist. Otherwise control cable routing is the same.

CONTROL COLUMN CABLE INSTALLATION

Aileron Control System Rigging
Figure 201 (Sheet 6 of 6)

AILERON - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

NOTE: ° The following procedures require that the hinge bolt access doors and the aileron control wing gap cover be removed.

° Removal and installation procedures for the left and right ailerons differ in that trim tab actuator and potentiometer wires are installed in the left aileron.

° When installing a new aileron it is necessary to trim both ends of the aileron as well as intall components from old aileron on the new aileron. Two aileron installation procedures have been provided.

° Each time an aileron is removed and installed, replace yoke bolts and center hinge bolts.

A. Remove Aileron (See figure 201.)

(1) Remove electrical power from aircraft.
(2) Remove hinge bolt access doors from underside of aileron and aileron control wing gap door from wing.
(3) Disconnect aileron trim tab actuator/potentiometer electrical wires at electrical wiring splice inside wing and pull electrical wiring out through aft wing spar.
(4) Disconnect balance tab push-pull tubes from wing bracket.
(5) Remove nut, washers, and clevis bolt connecting drive clevis to yoke. As bolt is removed, shims will be released. Save shims for reinstallation.

CAUTION: AILERON MUST BE SUPPORTED WHEN HINGE BOLTS ARE REMOVED OR INSTALLED. SIDE LOADS ON HINGE BEARINGS DURING REMOVAL AND INSTALLATION CAN DAMAGE HINGE BEARINGS.

(6) Remove hinge bolts that secure aileron to wing.
(7) Disconnect aileron bonding jumpers at wing and remove aileron.
(8) If aileron is to be replaced, remove yoke bolt, yoke, and shims from center hinge casting.

B. Install Existing Aileron (See figure 201.)

(1) If any portion of a control surface has been repainted, repaired, or replaced, a check must be made to determine whether balance is still within prescirbed limits. Refer to 27-00-00.

NOTE: Assure aileron hinge bearings rotate smoothly and are correctly installed before installing aileron.

(2) Align aileron on wing and connect aileron bonding jumper to aileron hinge fitting in wing. Secure aileron to wing with hinge bolts and safety wire bolts.
(3) Align aileron drive yoke with aileron drive clevis. Shim as necessary between yoke and drive clevis to prevent bearing preloading when clevis bolt is installed. Install clevis bolt, nut, and cotter pin.
(4) Route aileron trim tab actuator and potentiometer electrical wiring through wing ear spar and splice at bundle.
(5) Connect balance tab push-pull tubes to wing bracket.
(6) Check aileron for proper operation.
(7) Check aileron rigging. (Refer to Adjustment/Test, 27-10-00.)
(8) Install remaining access doors.
(9) Touch up paint.
(10) Restore electrical power to aircraft.

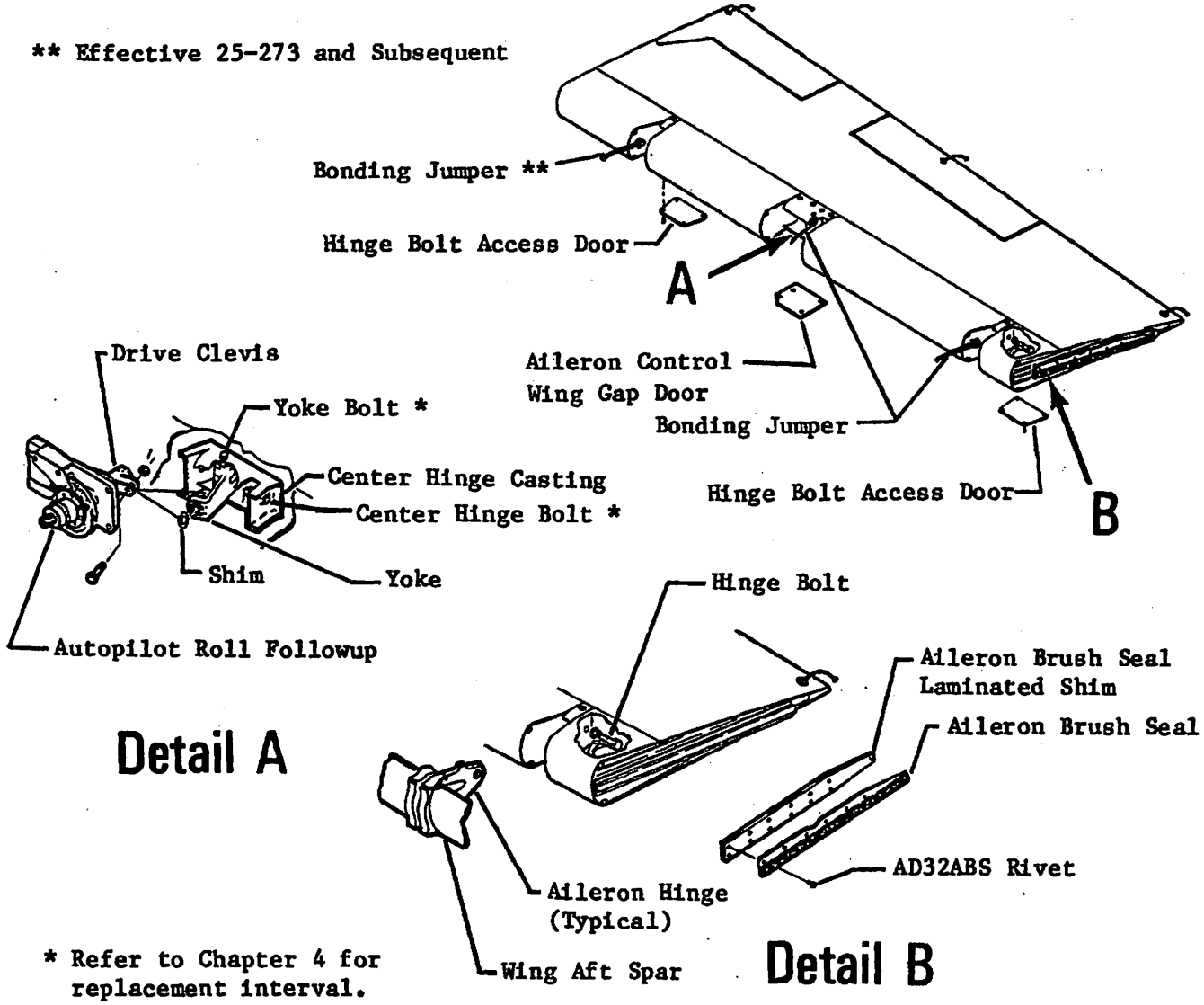
C. Install New Aileron (See figure 201.)

(1) Install yoke. Shim yoke top and bottom as necessary to assure that there is no end play of yoke in aileron and secure with yoke bolt, nut, and washer. Safety wire bolt head to center hinge casting.

CAUTION: AILERON MUST BE SUPPORTED WHEN HINGE BOLTS ARE REMOVED OR INSTALLED. SIDE LOADS ON HINGE BEARINGS DURING REMOVAL AND INSTALLATION CAN DAMAGE HINGE BEARINGS.

(2) Align aileron on wing hinge brackets and insert inboard, center, and outboard hinge bolts.
(3) Lay out trim lines on inboard and outboard ends of aileron as necessary to attain proper clearances as shown.
(4) Check all hinge points for proper alignment and freedom of movement.
(5) Remove aileron from wing and trim excess material from inboard and outboard ends as determined in step (3).
(6) Brush seals are temporarily installed. Install remaining rivets (P/N AD32ABS) securing brush seal to outboard rib.
(7) Install aileron hinge bolts and hinge bolt access doors.
(8) Mask off hinge and drive bearings. Paint aileron to match aircraft colors.
(9) Check aileron balance. (Refer to 27-00-00.)
(10) Apply a thin coat of Dow Corning No. 33 grease to aileron brush seals.
(11) Remove hinge bolt access doors and align aileron on wing. Secure aileron to wing with hinge bolts and safety wire bolts.
(12) Align yoke with drive clevis. Shim as necessary between yoke and drive clevis to prevent bearing preloading when clevis bolts is installed. Install clevis bolt, nut, and cotter pin.
(13) Route aileron trim tab actuator and potentiometer electrical wiring through wing rear spar and splice at bundle.
(14) Connect balance tab push-pull tubes to wing bracket.
(15) Check aileron for proper operation.
(16) Check aileron rigging. (Refer to Adjustment/Test, 27-10-00.)
(17) Install all remaining access doors.
(18) Restore electrical power to aircraft.

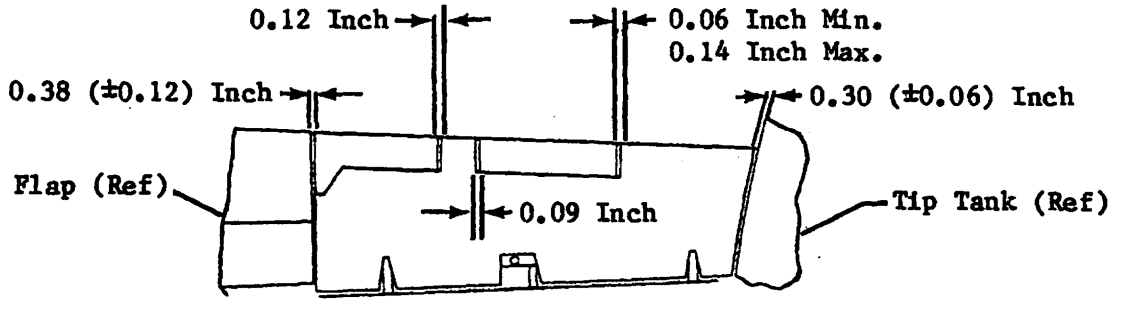
** Effective 25-273 and Subsequent



Detail A

Detail B

* Refer to Chapter 4 for replacement interval.



AILERON CLEARANCES

Aileron Installation
Figure 201



AILERON TRIM TAB AND ACTUATOR - MAINTENANCE PRACTICES

1. General
 A. The aileron trim tab is attached to the left aileron by a continuous-type hinge and is powered by a rotary-type electrical actuator located in the aileron leading edge.
 B. After a new trim tab has been installed, the aileron must be rebalanced. Refer to 27-00-00.
 C. Access to the trim tab actuator is through the trim tab actuator access cover on the aileron leading edge.

2. Removal/Installation
 A. Remove Aileron Trim Tab (See figure 201)
 (1) Remove aileron from wing. Refer to 27-10-01.
 (2) Disconnect trim tab actuator push-pull tube and bonding jumper from trim tab.
 (3) Using a No. 30 (0.128) drill, drill out rivets securing trim tab hinge to aileron spar. Remove trim tab from aileron.
 B. Install Aileron Trim Tab (See figure 201)
 (1) Install new trim tab on aileron with trim tab hinge between aileron rear spar flange and skin. Adjust trim tab for proper clearance between trim tab and aileron as shown.
 (2) Place a straightedge on aileron trailing edge and check to assure that trim tab trailing edge and aileron trailing edge are aligned.
 (3) With trim tab properly positioned, mark rivet locations on outboard and inboard ends of trim tab hinge. Remove trim tab and drill a No. 30 hole at each location mark.
 (4) Install trim tab and cleco into place. Using remaining tab hinge holes in aileron as a pattern, drill a No. 30 hole at each location except for the fourth hole from the outboard end of the hinge. Drill a No. 19 (0.166) hole at this location for bonding jumper attachment.
 (5) Secure trim tab to aileron with rivets (P/N MS20426AD4-6).
 (6) Connect trim tab bonding jumper and push-pull tube to trim tab.
 (7) Check aileron balance. Refer to 27-00-00.
 (8) Install aileron. Refer to 27-10-01.
 C. Remove Aileron Trim Tab Actuator (See figure 201)
 (1) Disconnect push-pull tube from aileron trim tab.
 (2) Remove trim tab actuator access cover from left aileron bottom leading edge.
 (3) Disconnect actuator electrical plug and slide plug back through spar.
 (4) Loosen potentiometer arm screw at potentiometer shaft and remove potentiometer arm from shaft.
 (5) Remove potentiometer and bracket from structure.

(6) Remove actuator attachment bolts and actuator with push-pull tube attached from aileron.
(7) Loosen control arm screw and remove actuator control arm from actuator.

D. Install Aileron Trim Tab Actuator (See figure 201)
(1) Place actuator control arm on actuator shaft with index spot on arm aligned with index line on actuator shaft. Tighten control arm screw.
(2) Install actuator with push-pull tube attached in aileron using attaching parts. Connect push-pull tube to trim tab.
(3) Install potentiometer and bracket using existing hardware. Tape trim tab in neutral position.
(4) Using an ohmmeter, rotate potentiometer shaft to center position (±10 ohms). Set potentiometer arm on potentiometer shaft and tighten arm screw.
(5) Connect electrical plug to actuator and install actuator access cover.

E. Remove Aileron Trim Tab Actuator Potentiometer (See figure 201)
(1) Remove trim tab actuator access cover from left aileron and remove arm from shaft.
(2) Loosen potentiometer arm screw at potentiometer shaft and remove arm from shaft.
(3) Remove potentiometer attachment nut and potentiometer from bracket.
(4) Disconnect electrical wiring at potentiometer and remove potentiometer from aileron.

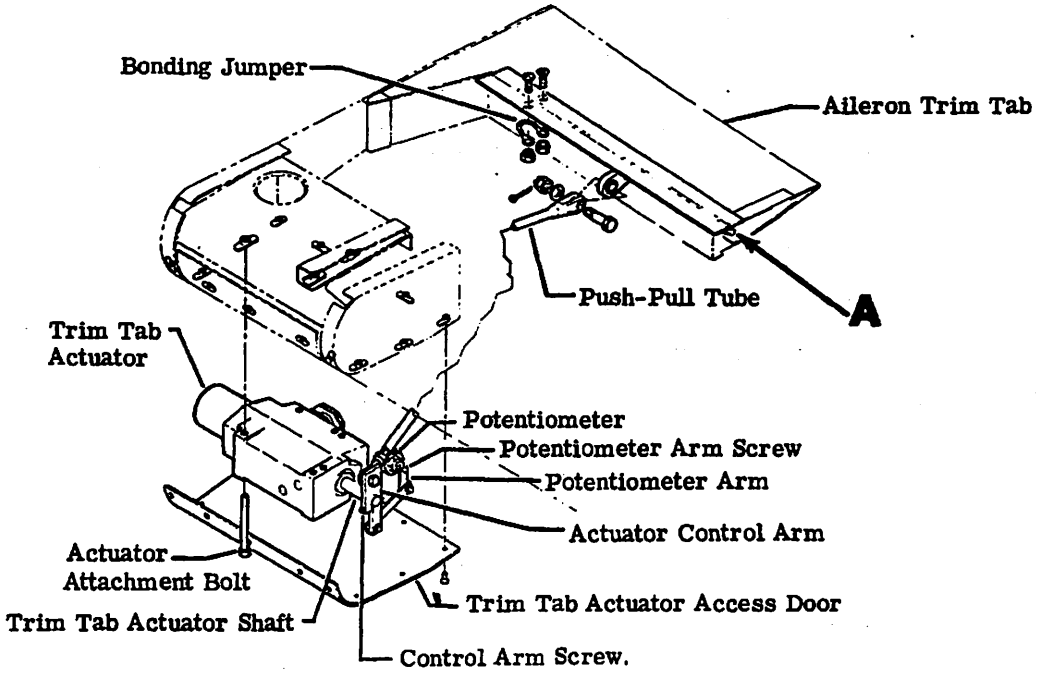
F. Install Aileron Trim Tab Actuator Potentiometer (See figure 201)
(1) Connect potentiometer electrical wiring to potentiometer.
(2) Install potentiometer on bracket using existing hardware. Tape trim tab in neutral position.
(3) Using an ohmmeter, rotate potentiometer shaft to center position (±10 ohms). Position potentiometer arm on potentiometer shaft and tighten arm screw.
(4) Check aileron trim tab for proper operation.
(5) Install trim tab actuator access cover.

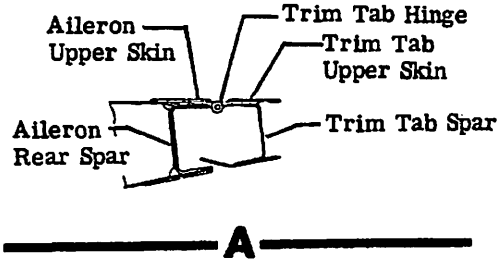
3. Adjustment/Test

A. Rig Aileron Trim Tab
(1) Remove trim tab actuator access cover on aileron leading edge.
(2) Actuate trim tab to streamline neutral position using the trim and trim arming switch on the control wheel.
(3) Attach clinometer to aileron trim tab.
(4) Check trim tab travel for 8° (+1°/-0°) both up and down. Adjust limit switches on trim tab actuator to obtain proper travel.

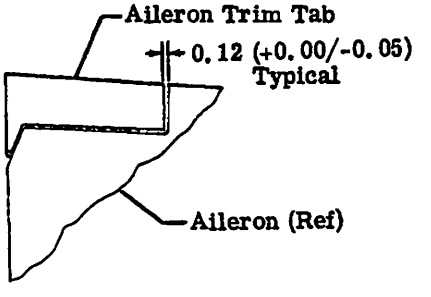
(5) Cycle trim tab several times checking for proper operation and correlation between aileron trim tab and aileron trim indicator on the cockpit center pedestal.
(6) Install trim tab actuator access cover.

4. Inspection/Check
A. Test Aileron Trim Tab for Proper Operation
(1) Press down the trim and trim arming switch, and set to LWD (left wing down) position and hold.
(2) The aileron trim tab will move down from the aileron in a smooth, responsive movement.
(3) The aileron trim indicator on the center pedestal will indicate corresponding trim tab travel.
(4) Press down the trim and trim arming switch, and set to RWD (right wing down) position and hold.
(5) The aileron trim tab will move up from the aileron in a smooth, responsive movement.
(6) The aileron trim indicator will indicate corresponding trim tab travel.





A



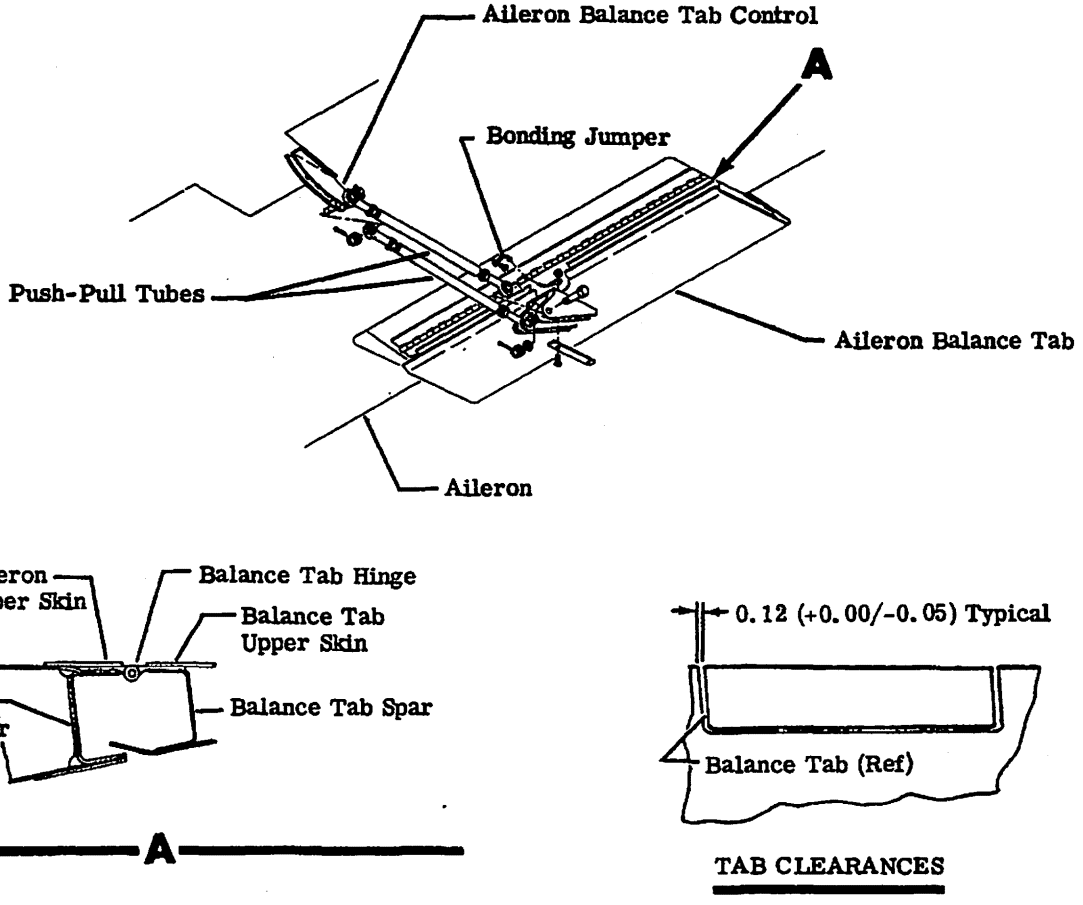
TAB CLEARANCES

Aileron Trim Tab Installation
Figure 201

AILERON BALANCE TAB - MAINTENANCE PRACTICES

1. General
 A. The aileron balance tabs are attached to the trailing edge of each aileron by a continuous-type hinge and are mechanically actuated by dual push-pull tubes connected to the fixed wing surface.

2. Removal/Installation
 A. Remove Aileron Balance Tab (See figure 201)
 (1) Remove aileron from wing. Refer to 27-10-01.
 (2) Disconnect balance tab push-pull tubes and bonding jumper from balance tab.
 (3) Using a No. 30 drill, drill out rivets securing balance tab hinge to aileron spar. Remove balance tab from aileron.
 B. Install Aileron Balance Tab (See figure 201)
 (1) Install new balance tab on aileron with balance tab hinge between aileron rear spar flange and skin. Adjust trim tab for proper clearance between balance tab and aileron as shown.
 (2) Place a straightedge on aileron trailing edge and check to assure that balance tab trailing edge and aileron trailing edge are aligned.
 (3) With balance tab properly positioned, mark rivet locations on outboard and inboard ends of balance tab hinge. Remove balance tab and drill a No. 30 hole at each location mark.
 (4) Install balance tab and cleco in place. Using remaining tab hinge holes in aileron as a pattern, drill a No. 30 hole at each location.
 (5) Secure balance tab to aileron with rivets (P/N MS20426AD4-6). Connect balance tab bonding jumper and push-pull tubes to balance tab.
 (6) Check aileron balance. Refer to 27-00-00.
 (7) Install aileron. Refer to 27-10-01.



Aileron Balance Tab Installation
Figure 201

AILERON TRIM POSITION INDICATOR - MAINTENANCE PRACTICES

1. Removal/Installation

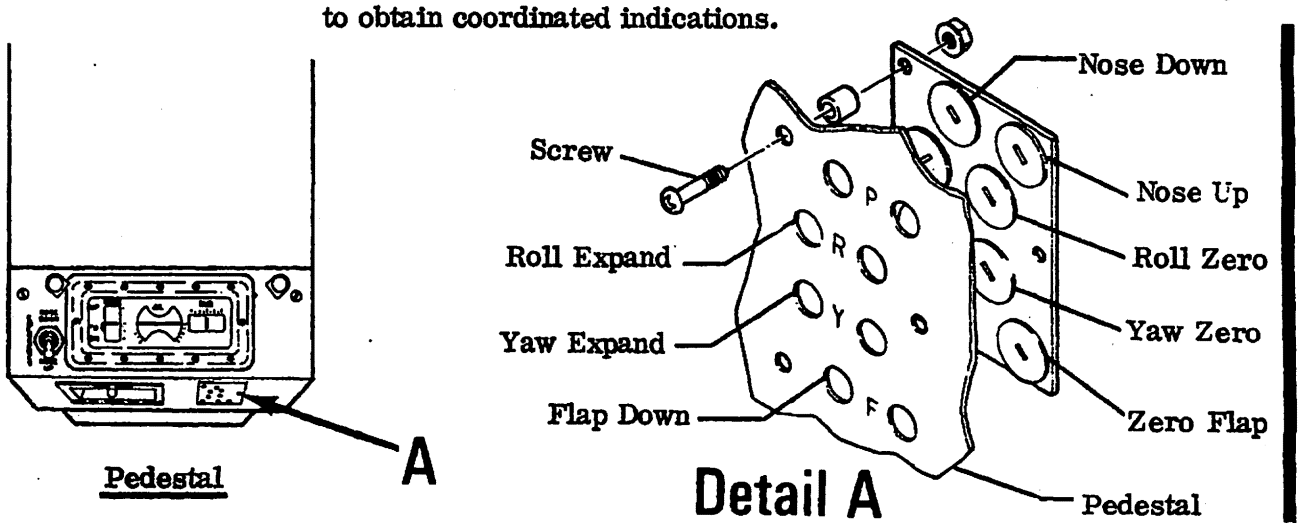
NOTE: Refer to 27-40-03 for removal and installation procedures for trim position indicator.

2. Adjustment/Test

A. Aileron Trim Position Indicator Calibration (See figure 201)

NOTE: Calibration of the aileron trim indicator is performed at the position indicator calibration assembly located inside the center pedestal.

(1) The aileron trim tab potentiometer must be approximately centered with tab in neutral position prior to calibrating aileron trim indicator.
 (a) Insert small screwdriver through holes in center pedestal and, using potentiometers, calibrate position indicator as follows.
 (b) Actuate aileron trim tab to neutral position.
 (c) Adjust Roll Zero potentiometer to obtain zero indication on roll trim indicator.
 (d) Actuate aileron trim tab to full up position.
 (e) Adjust Roll Expand potentiometer to obtain a right wing down indication on the aileron trim indicator.
 (f) Actuate aileron trim tab to full down position.
 (g) Adjust Roll Expand potentiometer to obtain left wing down indication on the aileron trim indicator.
 (h) Cycle aileron trim tab while checking respective indications on roll trim indicator. Readjust Roll Expand potentiometer as necessary to obtain coordinated indications.



Position Indicator Calibration Assembly
Figure 201

8-107A

CONTROL WHEEL - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Control Wheel (See figure 201)

NOTE: ● The control wheel assembly is only to be disassembled to the extent necessary to complete the desired repairs.

● Control wheel assemblies are to be inspected in accordance with current inspection intervals in Chapter 5.

(1) Remove electrical power from aircraft.
(2) Remove control wheel sector cover and control wheel escutcheon.
(3) Remove nut and washer which attach control wheel to shaft and pull control wheel from assembly.
(4) If control wheel is not to be removed further, allow control wheel to hang by electrical wiring and proceed to step (5). If control wheel is to be replaced, proceed as follows:
 (a) Disconnect plug (P80) forward of control column base.
 (b) Identify pins of plug (P80) which connect to control wheel to be replaced. (Refer to Wiring Manual.)
 (c) Remove applicable pins from plug (P80).
 (d) Pull control wheel away from control column, pulling electrical wiring through control column.
(5) Remove nut and washer that hold control wheel shaft in control column assembly.
(6) Remove adhesive from around phenolic washer on aft side of assembly and remove washer.
(7) Remove screws that attach aft bearing plate to control column and remove bearing plate.
(8) Remove floorboards and external access covers to gain access to aileron sector and remove forward aileron control cables from aileron sector.
(9) On Aircraft 25-067, 25-070 thru 25-186 NOT modified per SSK 947 (Replacement of Control Column Pulley and Shaft) or per AAK 76-4 (Reduced Approach Speed System), remove control wheel sector spring pin.
(10) Remove shaft and aft bearing from control column.
(11) On Aircraft 25-187 and Subsequent and prior Aircraft modified per SSK 947 (Replacement of Control Column Pulley and Shaft) or per AAK 76-4 (Reduced Approach Speed System), remove control wheel washers and spacers.
(12) Remove spring pins that hold control cables in place.
(13) Remove control cables, tagging cables by position for proper re-installation.

(14) Remove control wheel sector.
(15) Remove screws and nuts which hold forward bearing plate in place. Note position of screws. Larger screw is aileron stop and must be installed in proper position.
(16) Remove bearing plate.
(17) Press out forward bearing.

B. Install Control Wheel (See figures 201 and 202)

(1) Clean aft face of control column fork, aft bearing plate and attaching screws with MEK. Wipe dry with a clean dry cloth. Do not allow parts to air dry because MEK will leave a residue.

CAUTION: WHEN INSTALLING AFT BEARING, ENSURE THAT LOCTITE RETAINING COMPOUND IS NOT APPLIED TO INNER SURFACE OF BEARING.

(2) Apply a thin coating of Loctite Retaining Compound No. 680 to outer surface of aft bearing and mating surface of control column bearing support and insert bearing in aft control column bearing support. Wipe off excessive retaining compound with a rag moistened with trichlorethane. Allow retaining compound to cure for 1-4 hours at room temperature. Retaining compound can be force cured for 30 minutes at 180°F.
(3) After retaining compound has cured, install aft bearing plate and secure with attaching screws.

CAUTION: WHEN INSTALLING THE FORWARD BEARING, ENSURE THAT THE LOCTITE RETAINING COMPOUND IS NOT APPLIED TO THE INNER SURFACE OF THE BEARING.

(4) Apply a thin coating of Loctite Retaining Compound No. 680 to outer surface of forward bearing and mating surface of control column bearing support and insert bearing in forward control column bearing support. Wipe off excess retaining compound with a rag moistened with trichlorethane. Allow retaining compound to cure for 1-4 hours at room temperature. Retaining compound can be force cured for 30 minutes at 180°F.

CAUTION: ENSURE THAT AILERON STOP IS IN PROPER POSITION WHEN INSTALLED.

(5) After retaining compound has cured, install forward bearing plate and secure with attaching screws and nuts. (Aileron stop must be in proper position.)
(6) Prepare aft control column bearing support for installation of new phenolic washer as follows:

(a) Clean with MEK. Wipe dry with clean dry cloth. Do not allow to air dry because MEK will leave a residue.
(b) Repeat step (a) until all adhesive and contaminants are removed from mating surface.
(c) Brush a uniform coat of 1200 primer (product of Dow Corning Corp., Midland, Mich.) on mating surface. Allow primer to air-dry for 30 minutes.

(7) Apply a thin coat of RTV-108 adhesive (product of General Electric, Waterford, New York) to phenolic washer and install on control column bearing support. Ensure that phenolic washer is properly aligned. Allow adhesive to air-dry for 24 hours at room temperature.

(8) Install aileron control cables in control wheel sector and secure with spring pins.

(9) On Aircraft 25-187 and Subsequent and prior Aircraft modified per SSK 947 (Replacement of Control Column Pulley and Shaft) or per AAK 76-4 (Reduced Approach Speed System), install key in shaft, round side in toward center of shaft.

(10) On Aircraft 25-187 and Subsequent and prior Aircraft modified per SSK 947 (Replacement of Control Column Pulley and Shaft) or per AAK 76-4 (Reduced Approach Speed System), position washers and spacers inside control column bearing support.

(11) Insert shaft so that index mark on splined shaft is at the 12 o'clock position and secure on forward side of control column with washer and nut.

(12) On Aircraft 25-061, 25-070 thru 25-186 NOT modified per SSK 947 (Replacement of Control Column Pulley and Shaft) or AAK 76-4 (Reduced Approach Speed System), install coiled spring pin in control wheel sector.

(13) If control wheel has not been removed from assembly, proceed with step (14). If control wheel has been removed from control column assembly, proceed as follows:
(a) Route electrical wiring down through control column.
(b) Install electrical pins into plug (P80). (Refer to Wiring Manual.)
(c) Connect plug (P80) to receptacle (J80), located forward of control column base.

(14) When installing control wheel, loop wiring approximately 1-1/2 turns outside control wheel hub, then 1-1/2 turns inside control wheel hub. Install control wheel and assure that control wheel is in neutral position (horizontal with ailerons faired).

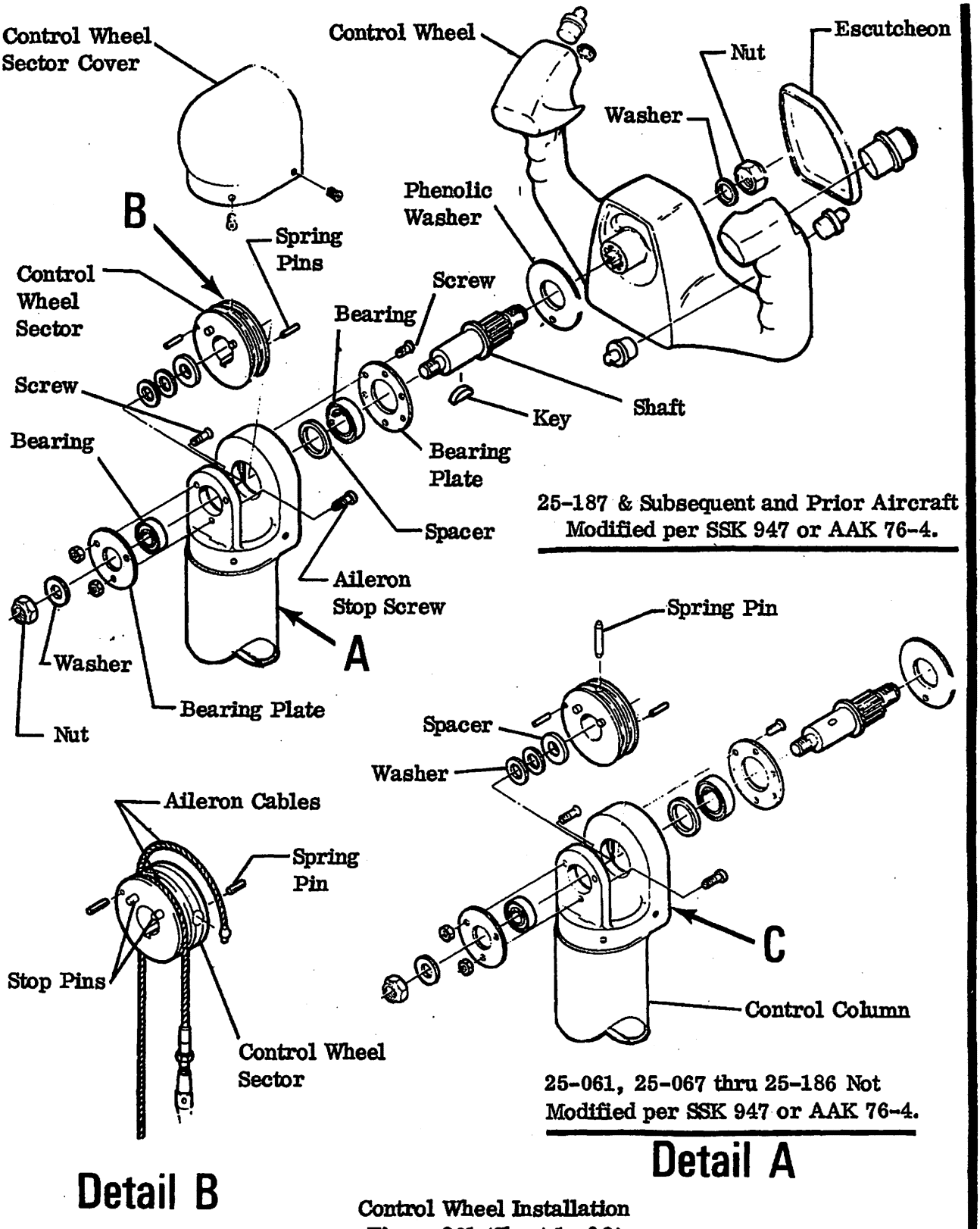
(15) Secure control wheel with washer and nut. Install control wheel escutcheon.

(16) Install aileron control cables on aileron sector.

(17) Rig and inspect aileron control system per 27-10-00.

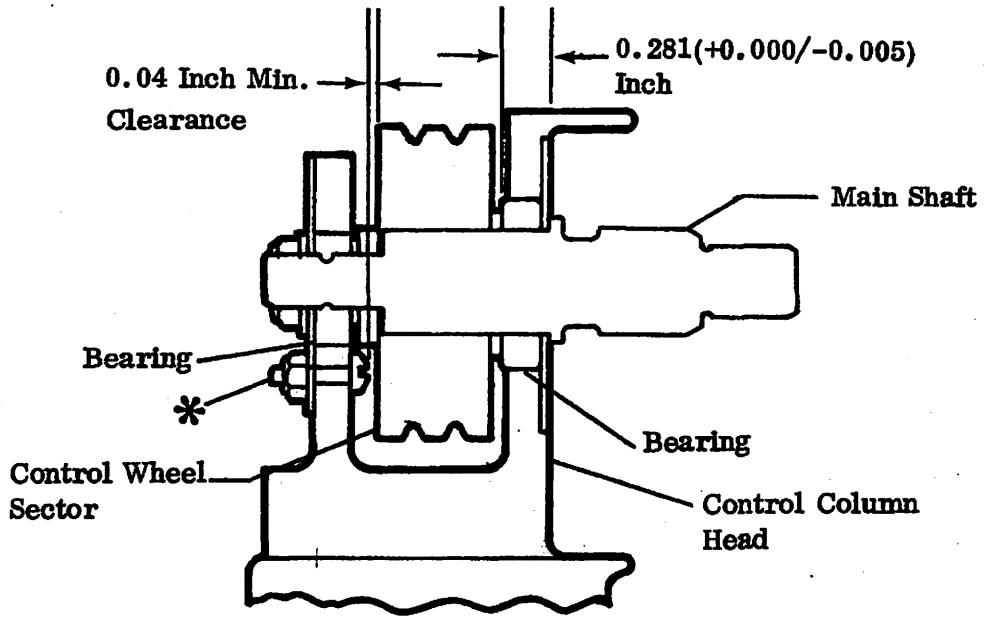
(18) Check that control wheels are synchronized in their movement and do not move past control column aileron stop.

(19) Restore electrical power to aircraft.
(20) Test all functions of control wheel, i.e., microphone, nose steering, etc. to ensure that all electrical connections are installed correctly.
(21) Install all access panels and floorboards removed.



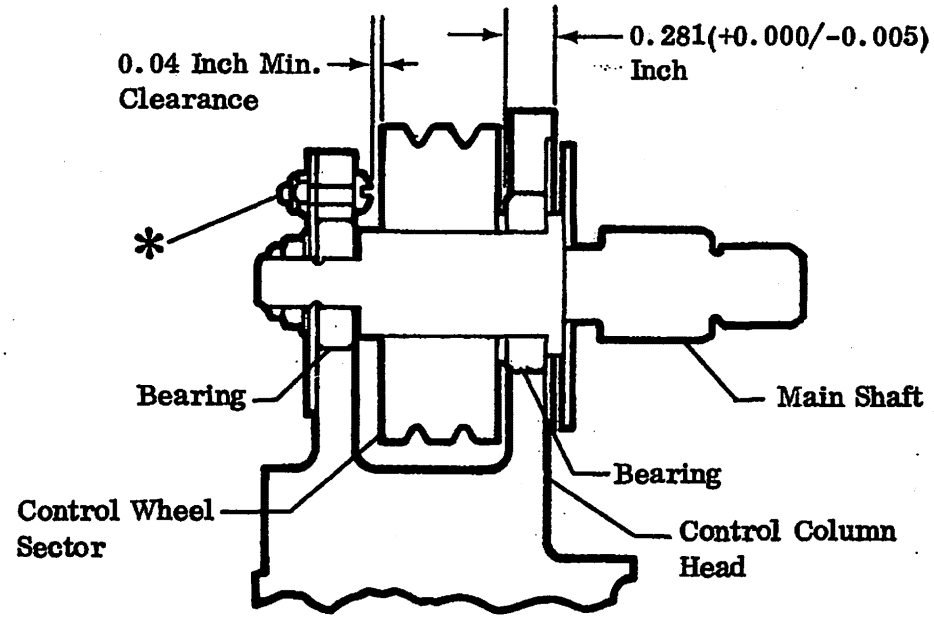
Control Wheel Installation
Figure 201 (Sheet 1 of 2)

8-64B



Aircraft 25-061, 25-070 thru 25-081 not modified per SSK 947 (Replacement of Control Column Pulley and Shaft) or AAK 76-4 (Reduced Approach Speed).

Detail C



Aircraft 25-082 and Subsequent and Prior Aircraft modified per SSK 947 or AAK 76-4.

*File end of stop screw if necessary to provide minimum clearance.

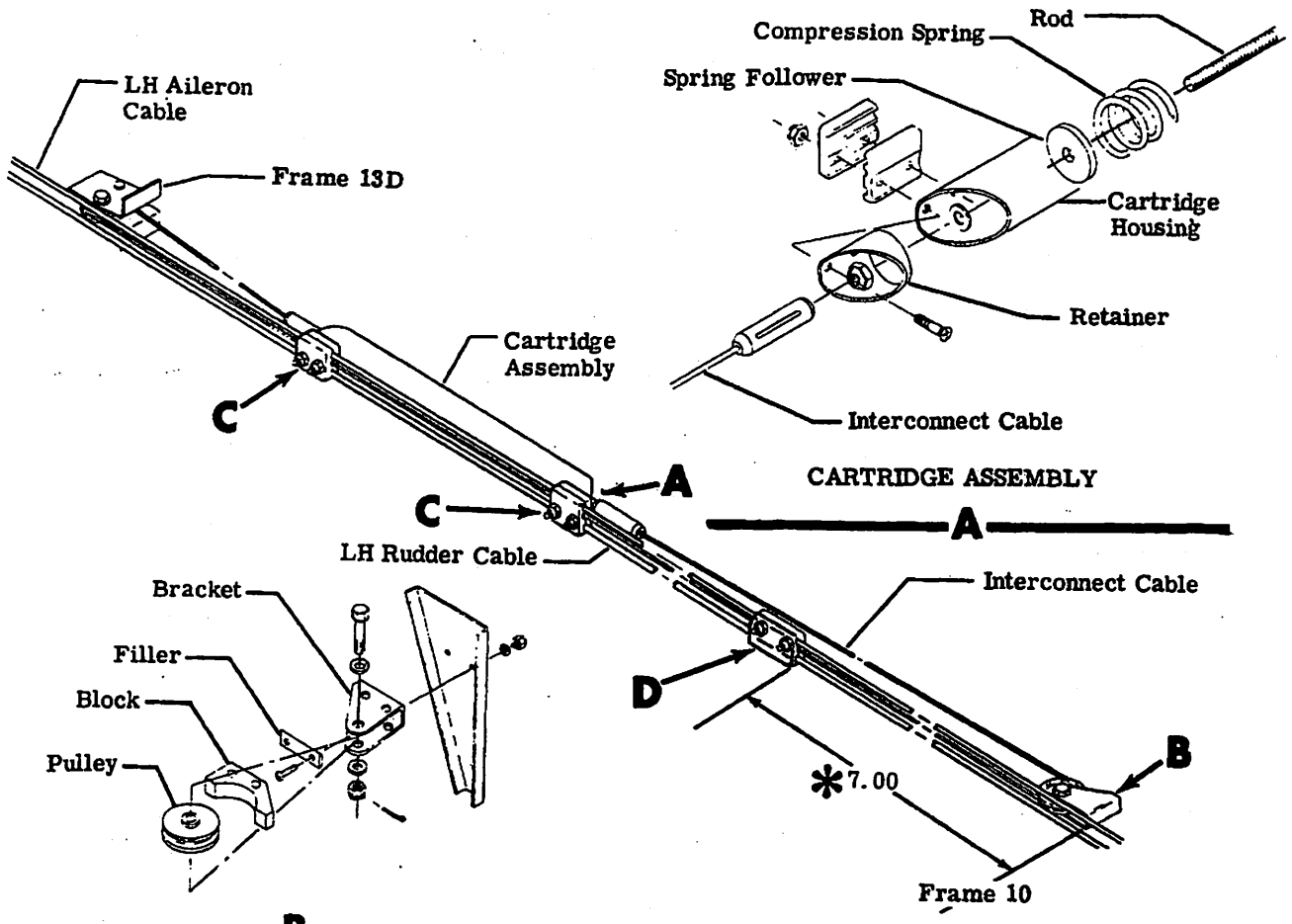
Detail C

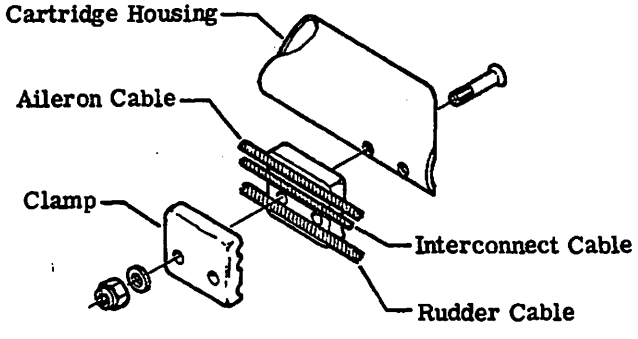
Control Wheel Installation
Figure 201 (Sheet 2 of 2)

AILERON-RUDDER INTERCONNECT - MAINTENANCE PRACTICES

1. General
 A. The aileron-rudder interconnect is installed from frame 10 to frame 13D on the LH side of the cabin.

2. Adjustment/Test
 A. Rig Aileron-Rudder Interconnect (See figure 201)
 (1) Remove floorboards necessary to gain access to aileron rudder interconnect.
 (2) Adjust nuts on rod of spring cartridge so they contact the spring followers and the spring followers contact the retainers simultaneously on both ends. Maximum end play shall not exceed 0.010 inch.
 (3) With both ailerons and the rudder in neutral, adjust the interconnect cable assembly to 25 (±5) pounds at 75°F. Refer to 27-00-00 for cable tension temperature-load correction.
 (4) Operate rudder and aileron systems through full travel in all combinations and check for full freedom of movement. Left rudder should produce left aileron and right rudder should produce right aileron.
 (5) Install floorboards.

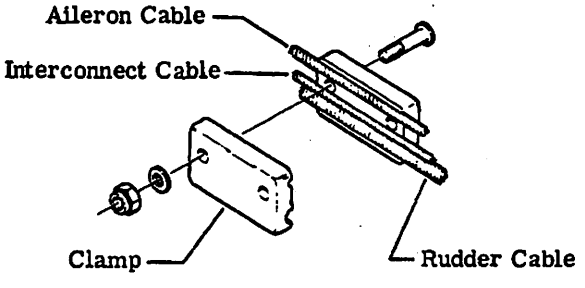




Assure that ailerons and interconnect cables are routed through clearance holes and that rudder cable is secure in this clamp.

TYPICAL CABLE CLAMP

C



Assure that rudder cable is routed through clearance hole and that aileron and interconnect cables are secure in this clamp.

D

Aileron-Rudder Interconnect
Figure 201

RUDDER AND TAB - DESCRIPTION AND OPERATION

1. Description

A. Directional control of the aircraft is provided by the rudder. The rudder system consists of a dual set of rudder pedals, a cable system, and rudder assembly. The rudder, mounted on a torque tube is connected to the vertical stabilizer at three hinge points. Rudder control is provided manually by either set of rudder pedals or automatically through the primary or secondary yaw damper servo actuator.

B. A rudder trim tab system is an integral part of the rudder and further increases directional stability of the aircraft. The rudder trim tab is attached to the rudder by a continuous-type hinge and is powered by a rotary-type electrical actuator located in the lower leading edge of the rudder. The Yaw Trim Switch on the pedestal controls the tab actuator and a rudder trim indicator indicates trim tab travel. Trim tab travel is 15° (±1°) left and right.

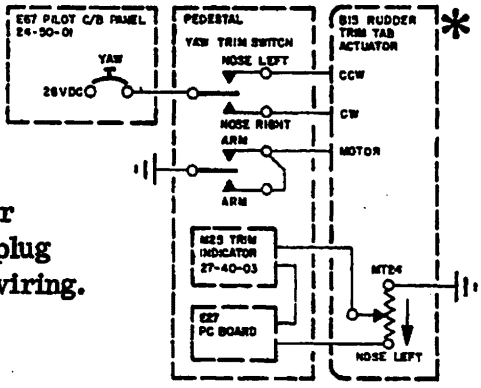
C. The rudder pedals operate the brakes and nose wheel steering as well as the rudder. For brake system information refer to Chapter 32 and for steering system information refer to Chapter 32.

D. The rudder trim position indicator, located on the cockpit center pedestal, indicates rudder trim tab travel. Indicator input is from a potentiometer attached to the rudder trim tab actuator.

E. Calibration of the rudder trim indicator is performed at the position indicator calibration assembly located in the center pedestal.

2. Operation

A. Rudder control motion is transferred from interconnected rudder control pedals through push-pull tubes, a bellcrank assembly, then a closed loop drive cable assembly to a drive bellcrank connected to the rudder torque tube. Two sets of rudder travel stop bolts (primary and secondary) are installed. The primary stop bolts, located in the stinger, limit rudder travel when contacted by the rudder torque tube bellcrank. The secondary stop bolts, located on the floorboards forward of the rudder pedals, limit rudder travel when contacted by the rudder pedal arms.

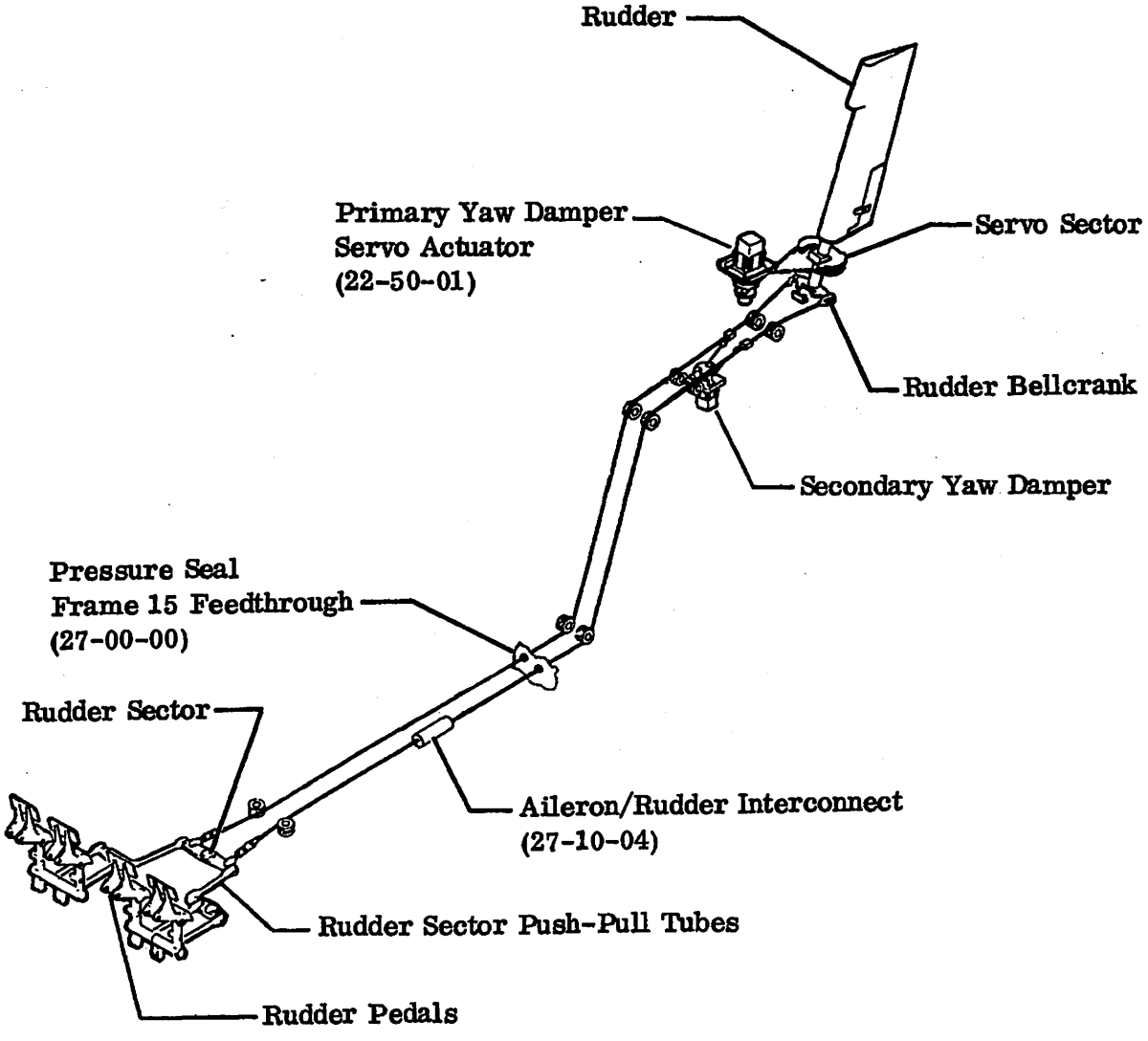


Rudder Trim Tab Electrical Control and Indicating Schematic
Figure 1

* The trim tab actuator either incorporates an electrical plug or is spliced into aircraft wiring.

B. Rudder trim tab motion is controlled by the yaw trim switch. Setting the switch to LEFT or RIGHT energizes the trim tab actuator to either clockwise or counterclockwise rotation. A push-pull tube connects the actuator to the trim tab. Maximum trim tab travel is 15° (±1°) both left and right.

C. The rudder trim tab may be manually operated by using the Rudder Trim Switch on the pedestal. A trim tab position indicator is also installed on the pedestal.



Rudder Control System Installation
Figure 2

8-22C-1

RUDDER AND TAB CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Adjustment/Test

A. Tools and Equipment required for Adjustment/Test

NAME	NUMBER	MANUFACTURER	USE
Rudder and Tab Rigging Fixture	2471005	Gates Learjet Corp.	Measure degree of control deflection.
Tensiometer	2471007	Gates Learjet Corp.	Measure cable tension.

B. Rig Rudder Control System (See figure 201)

(1) Remove access covers and floorboards as required for access to rudder control system.

(2) Insert rigging pin in left and right rudder pedal bellcranks. This provides a stationary point from which rudder control system can be rigged.

(3) Adjust rudder pedal bellcrank push-pull tubes until centerline of rudder pedal supports are 7° (±1°) aft of vertical.

(4) Adjust rudder sector push-pull tubes until rudder sector bellcrank is perpendicular to BL 0.

(5) Determine that rudder is in neutral position by use of contour plates or rudder and tab rigging fixture (P/N 2471005). If rudder is not in neutral, adjust cable turnbuckles until rudder is in neutral.

NOTE: When cable turnbuckles are loosened or tightened to adjust cable tension, it may be necessary to readjust rudder to neutral position. Repeat cable adjustments alternately until rudder is rigged to neutral at the required cable tension.

(6) Adjust turnbuckles to obtain cable tension of 75 (±5) pounds at 75° F. (Refer to 27-00-00 for cable tension temperature-load correction.)

(7) Remove rigging pin.

(8) Using rudder and tab rigging fixture (P/N 2471005), or see figure 201 if fixture is not available, adjust the primary travel stops in the tailcone to obtain a rudder travel as shown in figure 201.

(9) Lock primary stop bolts with jamnuts. On Aircraft 25-198 and Subsequent and Aircraft modified by SB 23/24/25-266, safety wire stop bolts.

(10) Adjust the secondary travel stops to contact when 100 (±25) pounds of pressure is applied to the rudder pedals. Lock stop bolts with jamnuts. On Aircraft 25-198 and Subsequent and Aircraft modified by SB 23/24/25-266, safety wire stop bolts.

(11) Adjust yaw damper servo cable tension to 25 (±5) pounds at 75° F. (Refer to 27-00-00 for cable tension temperature load correction.)
(12) Operate rudder several times to assure proper travel and freedom of movement.
(13) Perform Rudder Control System Friction Check.
(14) Check capstan slip clutch adjustment. (Refer to Chapter 22.)
(15) Install and secure all access covers and floorboards prior to flying aircraft.

2. Inspection/Check

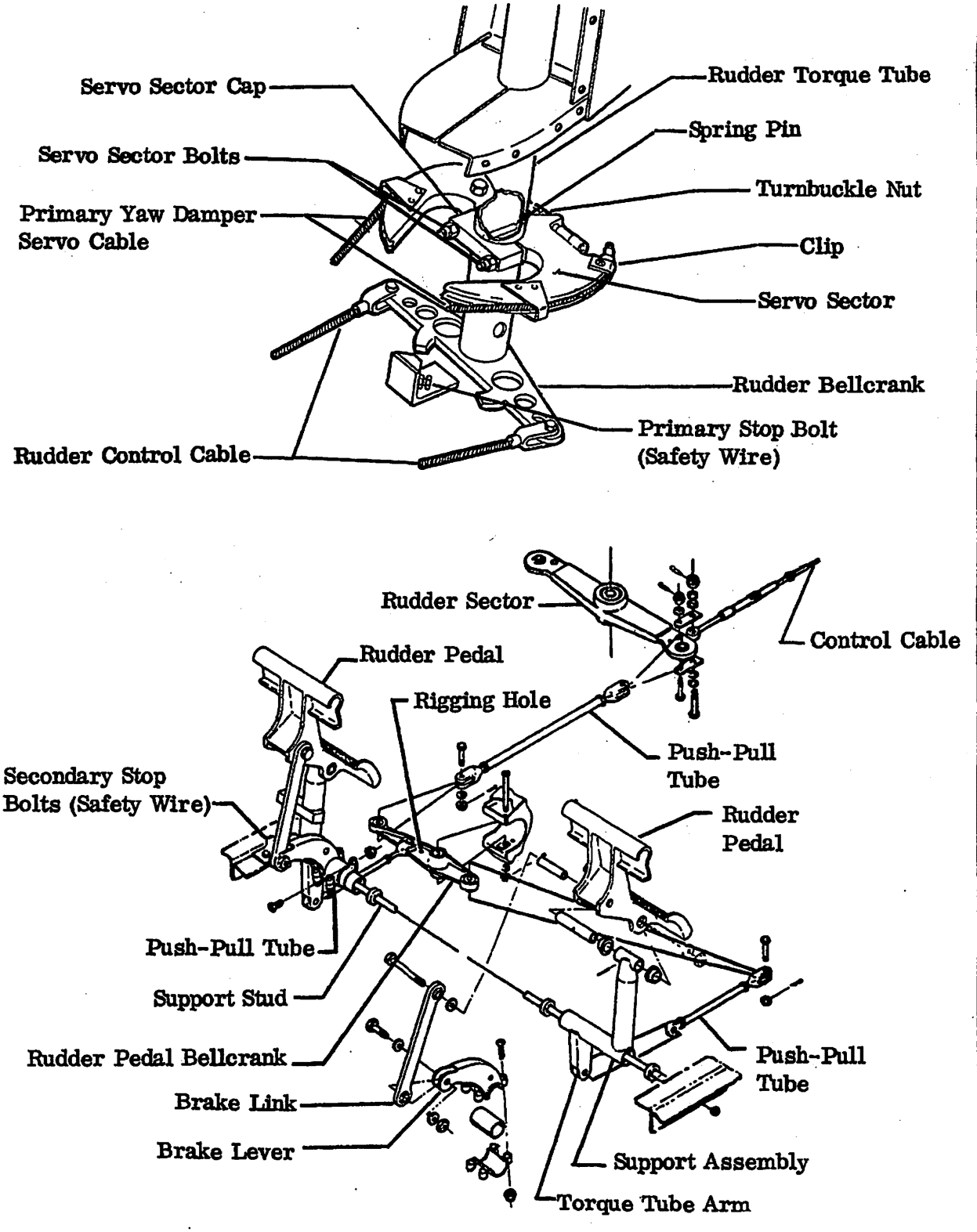
A. Check Rudder for Proper Operation

(1) During operational check, observe response and freedom of movement of the rudder pedals.
(2) Apply force to pilot's outboard rudder pedal; observe rudder trailing edge movement to the left.
(3) Apply force to pilot's inboard rudder pedal; observe rudder trailing edge movement to the right.
(4) Relax force applied to rudder pedals; observe rudder movement to the neutral position.
(5) Observe response and freedom of movement of the rudder pedals.
(6) Apply force to copilot's outboard rudder pedal; observe rudder trailing edge movement to the right.
(7) Apply force to copilot's inboard rudder pedal; observe rudder trailing edge movement to the left.
(8) Relax force applied to rudder pedals; observe rudder movement to the neutral position.

B. Perform Rudder Control System Friction Check

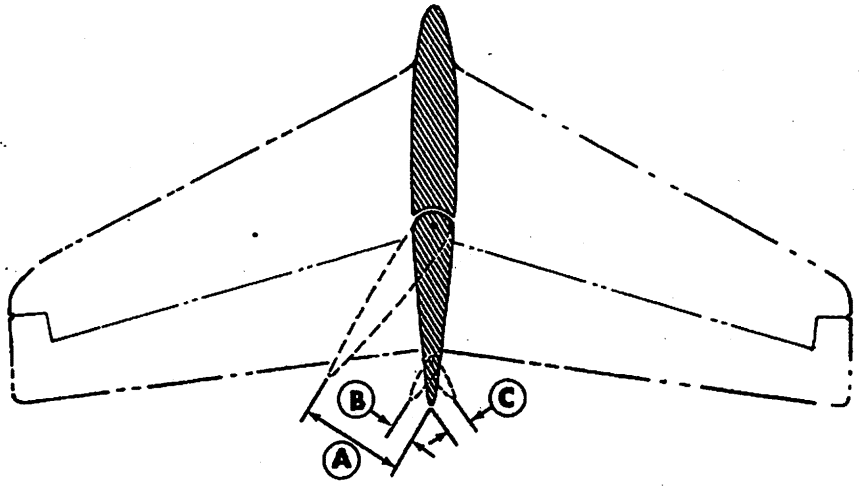
NOTE: Autopilot is to be disengaged when this check is made.

(1) With aileron and rudder systems centered, place a push-pull spring scale (accurate within 1/2 pound) at lower aft corner of rudder trim tab.
(2) Slowly apply a horizontal force to the rudder trim tab.
(3) When rudder starts its normal movement, read force applied. Maximum allowable force is 6.5 pounds.
(4) If force applied is greater than 6.5 pounds, check for proper cable tensions; check pulleys, sector and bellcrank for freedom of movement; inspect system for binding; check system for proper lubrication; and ensure that aileron and rudder control systems are rigged properly.
(5) After all discrepancies have been corrected, perform steps (1) through (3) and ensure that force applied is less than 6.5 pounds.



8-20C
8-69C

Rudder and Trim Tab Rigging
Figure 201 (Sheet 1 of 2)



RUDDER TRAVEL CHECK:

1. Set rudder trim tab to neutral (streamlined with rudder).
2. Set rudder to neutral (align lower tip of rudder trim tab with point of stinger).
3. Press left rudder pedal until rudder horn contacts rudder stop bolt, then measure dimension A (distance between lower tip of trim tab and stinger point).
4. Dimension A should be between 10.74 to 11.83 inches (30° + 2°, -1°).
5. Repeat procedure for right rudder travel.

RUDDER TRIM TAB TRAVEL CHECK:

1. Set rudder trim tab to neutral (streamlined with rudder).
2. Set rudder to neutral (align lower tip of rudder trim tab with point of stinger) and block rudder to hold this position.

NOTE: Since the rudder trim tab is not hinged on the centerline of the aircraft, a differential travel is required to obtain 15° (±1°) relative to the centerline of the aircraft.

3. Operate yaw trim switch to obtain full left travel, then measure dimension B between lower tip of trim tab and stinger point.
4. Dimension B should be between 1.31 and 1.51 inches.
5. Operate trim tab to full right travel, then measure dimension C.
6. Dimension C should be between 1.28 to 1.42 inches.

Rudder and Trim Tab Rigging
Figure 201 (Sheet 2 of 2)

RUDDER - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: Access to the torque tube hinge bolt is through access covers on either side of the tailcone stinger. Access to the upper and center rudder hinge points is through access covers on the rudder leading edge.

A. Remove Rudder (See figure 201)

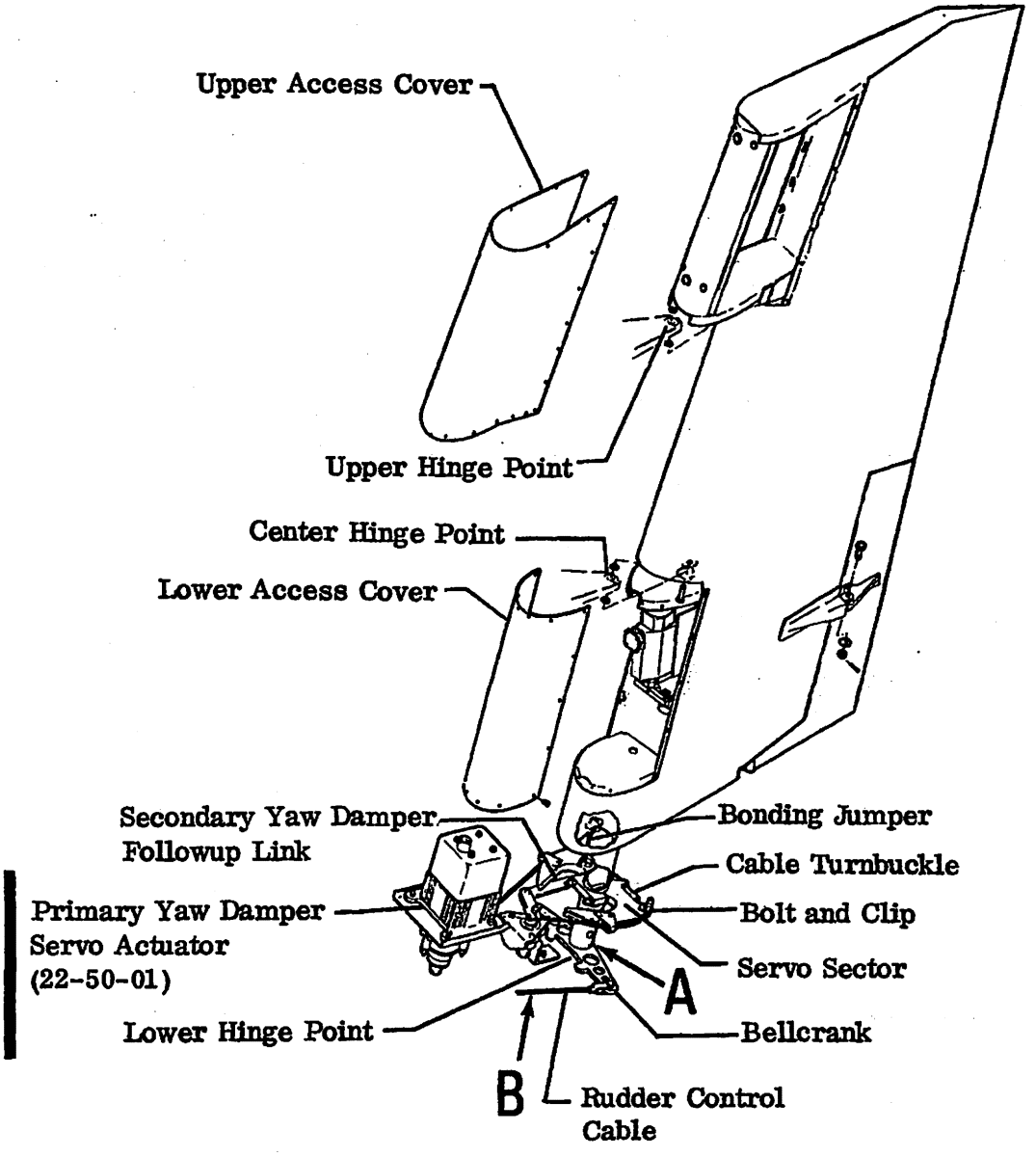
(1) Remove screws from upper and lower access covers on rudder leading edge.
(2) Deflect rudder to one side and remove access covers.
(3) Remove rudder servo sector and bellcrank access covers from tailcone.
(4) Disconnect bonding jumper from rudder.
(5) Disconnect trim tab actuator electrical plug and remove wire bundle clamp from rib.
(6) Remove bolt and clip from the left side of the servo sector and remove cable.
(7) Remove the two nuts on the forward side of the sector which hold the sector to the torque tube. Remove sector with turnbuckle end of cable attached to sector.

CAUTION: WHEN CLAMPING RUDDER CABLES DO NOT OVER-TIGHTEN CLAMPS. OVERTIGHTENING CLAMPS COULD CAUSE CABLE DAMAGE.

(8) Clamp the rudder control cables where they pass through the bulkhead in a manner such that cables will remain tight forward of the bulkhead (see detail B).
(9) Deflect rudder so that one of the rudder control cables becomes slack, remove that bolt and cable end from bellcrank. Remove opposite cable in the same manner.
(10) Remove torque tube hinge fitting bolt at the bottom of the rudder torque tube.
(11) Remove center and upper rudder hinge bolts and washers.
(12) Deflect rudder so that it clears the aft bullet navigation light fairing and remove rudder.

B. Install Rudder (See figure 201)

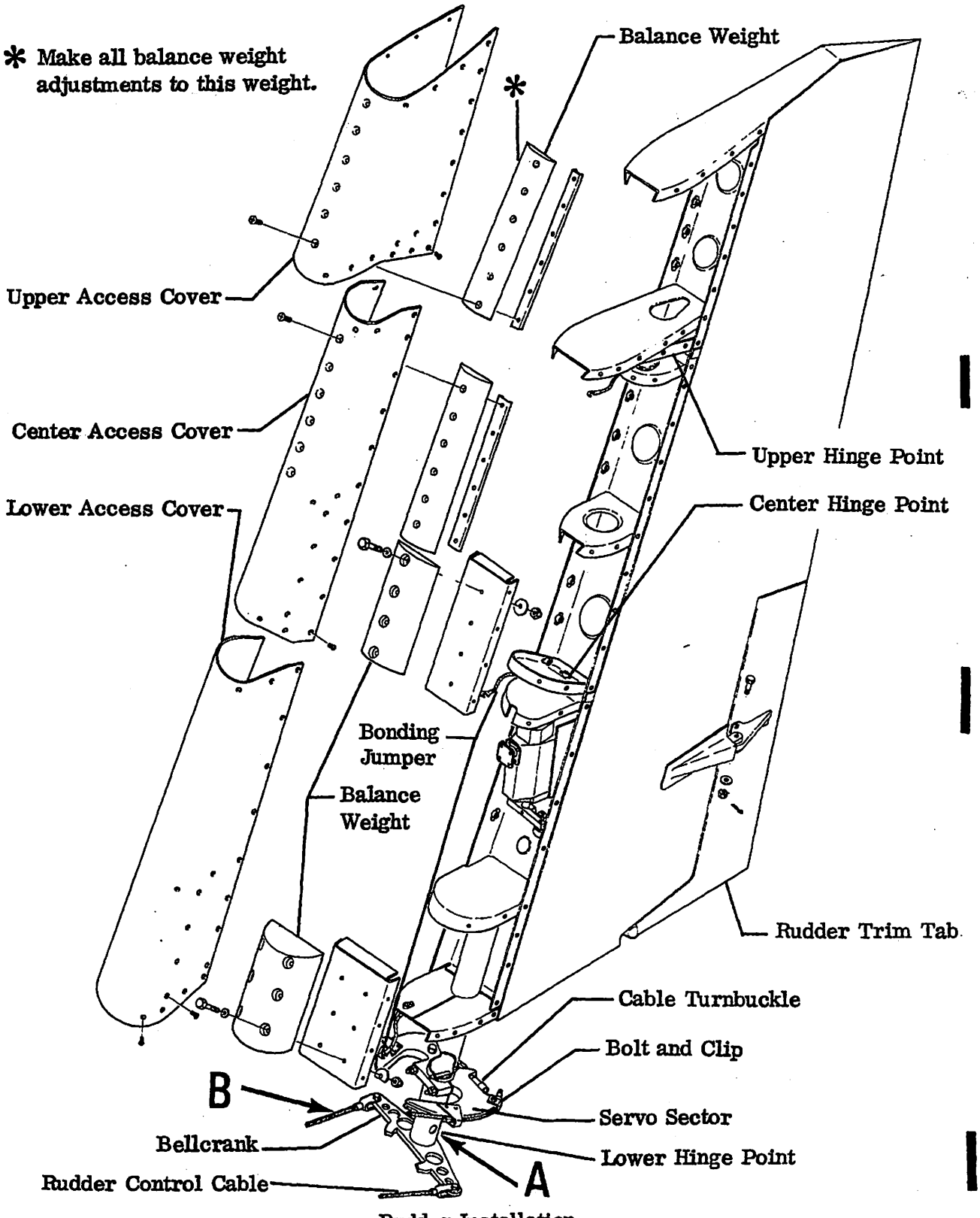
NOTE: If any portion of a control surface is repainted, repaired, or replaced, a check must be made to determine if the balance is still within the prescribed limits. Refer to 27-00-00.



Rudder Installation
Figure 201 (Sheet 1 of 3)

6-8D

EFFECTIVITY: Aircraft 25-061, 070 thru 25-205 not Equipped with Reduced Approach Speed System.

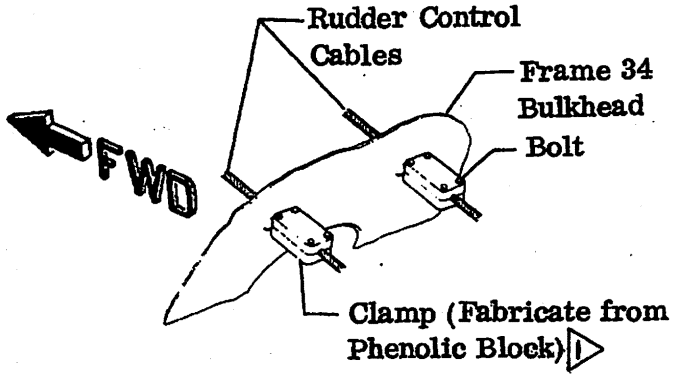


Rudder Installation
Figure 201 (Sheet 2 of 3)

5-13D

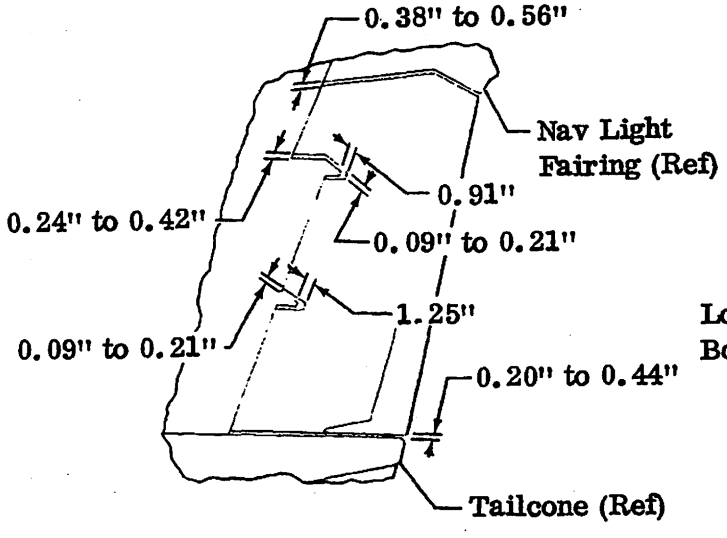
EFFECTIVITY: 25-206 and Subsequent and Prior Aircraft equipped with Reduced Approach Speed System

CAUTION: OVERTIGHTENING OF CLAMP COULD CAUSE CABLE DAMAGE.

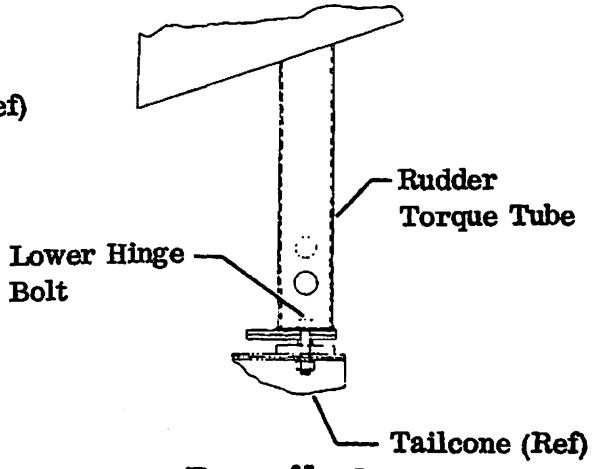


RUDDER CONTROL CABLE CLAMPING

Detail B



RUDDER CLEARANCES.



Detail A

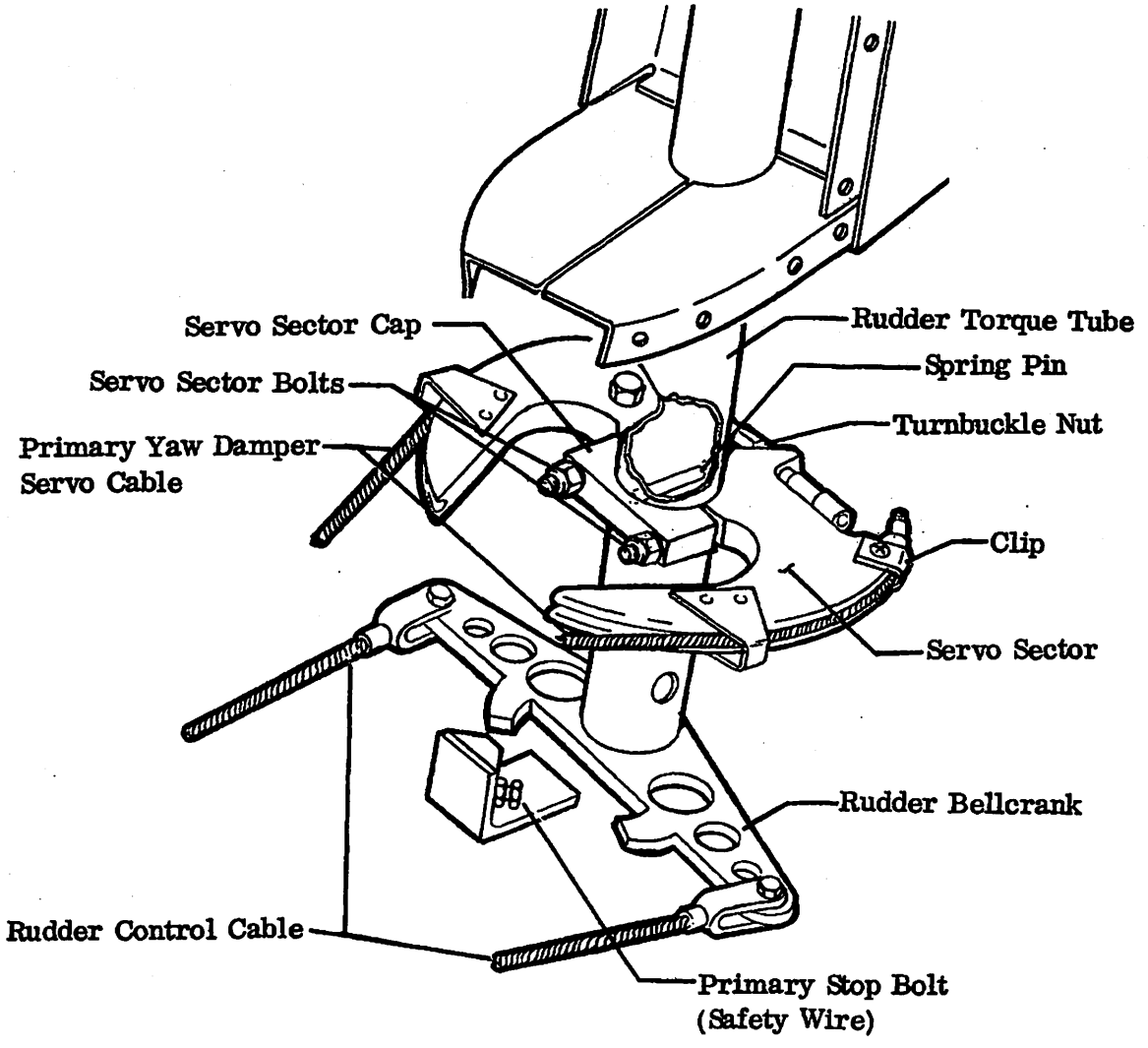
Rudder Installation
Figure 201 (Sheet 3 of 3)

(1) Position rudder on vertical stabilizer.
(2) Insert washers and bolts at the upper and center hinge points.
(3) Attach torque tube to lower hinge fitting point and attach bonding jumper. Assure that rudder clears vertical stabilizer skin by 0.30 (+0.05/-0.18) inch.
(4) Deflect rudder to one side and attach rudder control cable on that side. Attach opposite cable in same manner. Remove clamps from rudder control cables.
(5) Attach servo sector to torque tube. Attach cable end to servo sector with bolt and clip.
(6) Adjust servo cable to 25 (±5) pounds at 75° F. (Refer to 27-00-00 for cable tension temperature load correction.)
(7) Adjust rudder control cables to 75 (±5) pounds at 75° F. Refer to 27-00-00 for cable tension temperature load correction.
(8) Assure that trim tab actuator wire bundle is wrapped approximately 1-1/2 turns around the torque tube and tied so that the wire will not fall into the bellcrank stops.
(9) Connect trim tab actuator electrical connector and secure wire bundle clamp to rib.
(10) Check rudder control system for proper rigging.
(11) Install upper and lower access covers.

RUDDER SERVO SECTOR - MAINTENANCE PRACTICES

1. General
 A. The autopilot primary yaw damper servo is connected to the rudder servo sector.
 B. Removal and installation of the rudder servo sector is necessary when a new rudder is to be installed.

2. Removal/Installation
 A. Remove Rudder Servo Sector (See figure 201)
 (1) Remove floorboards as required to gain access to rudder pedal bellcrank. Insert rigging pin in rudder pedal bellcrank.
 (2) Remove rudder servo sector and bellcrank access cover from tailcone.
 (3) Remove primary yaw damper servo cable turnbuckle nut, clip, and cotter pin, releasing yaw damper servo cable ends from rudder servo sector.
 (4) Remove nuts and bolts from servo sector, releasing servo sector and cap from rudder torque tube.
 B. Install Rudder Servo Sector (See figure 201)
 (1) Align servo sector and cap with servo sector spring pin in rudder torque tube. Secure servo sector and cap on torque tube using existing hardware.
 (2) Attach primary yaw damper servo cable to servo sector using clip, nut, and cotter pin.
 (3) Adjust servo cable to 25 (±5) pounds at 75°F. Refer to 27-00-00 for cable tension temperature load correction.
 (4) Install rudder servo sector and bellcrank access cover on tailcone.
 (5) Remove rigging pin from rudder pedal bellcrank and install floorboards.

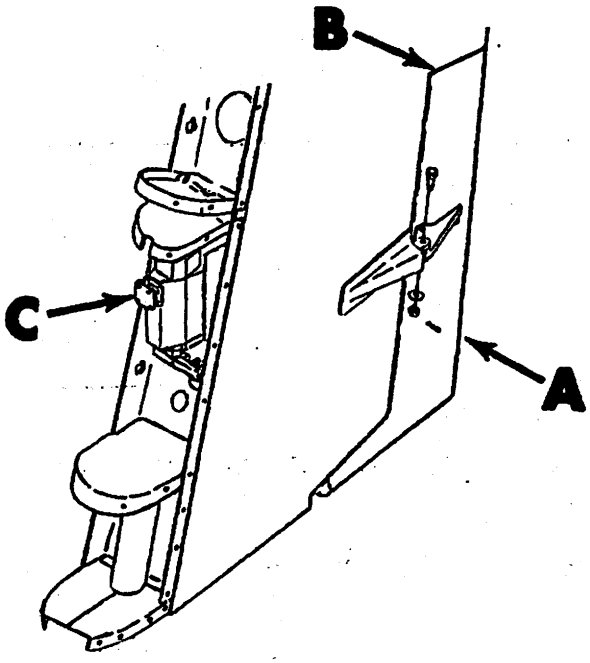


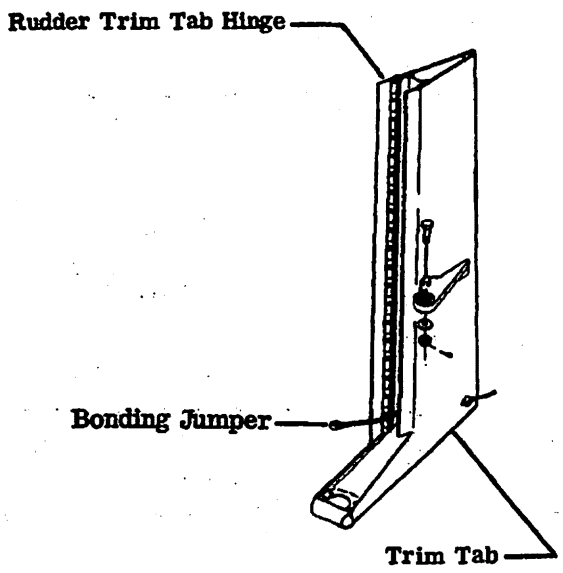
Rudder Servo Sector Installation
Figure 201

RUDDER TRIM TAB AND ACTUATOR - MAINTENANCE PRACTICES

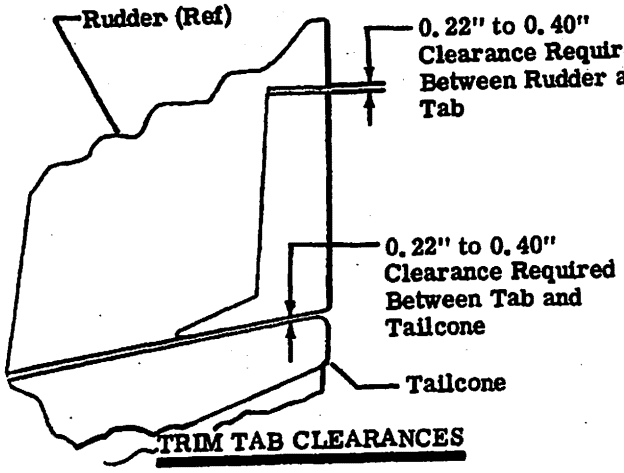
1. General
 A. The rudder trim tab is a small airfoil attached to the trailing edge of the rudder. The rudder trim tab actuator is installed in the rudder leading edge.
 B. Access to the rudder trim tab actuator is through the lower access cover on the rudder leading edge.

2. Removal/Installation
 A. Remove Rudder Trim Tab (See figure 201)
 (1) Remove rudder from aircraft. . Refer to 27-20-01.
 (2) Disconnect trim tab actuator push-pull tube and bonding jumper from trim tab and trim tab actuator.
 (3) Using a No. 30 (0.128) drill, drill out tab hinge rivets from rudder. Remove rudder trim tab from rudder.
 B. Install Rudder Trim Tab (See figure 201)
 (1) Install trim tab on rudder with trim tab hinge between rudder spar flange and skin of rudder. Adjust trim tab for proper clearances between rudder tab and rudder as shown.
 (2) Place a straightedge on rudder trailing edge and adjust rudder trim tab until both the rudder and rudder trim tab trailing edges are aligned.
 (3) With trim tab properly positioned, mark rivet locations at upper and lower ends of trim tab. Remove trim tab from rudder and drill a No. 30 hole at each location. Install trim tab and cleco in place. Using remaining holes in rudder as a pattern, drill a No. 30 hole at each location except the second from the bottom. Drill a No. 19 (0.166) hole at this location for bonding jumper attachment. Secure trim tab to rudder with rivets (P/N MS20426AD4) and connect bonding jumper to trim tab. Install push-pull tube.
 (4) Install rudder. Refer to 27-20-01.
 C. Remove Rudder Trim Tab Actuator (See figure 201)
 (1) Remove lower access cover from rudder leading edge.
 (2) Remove rudder bellcrank access cover from top of tailcone.
 (3) Disconnect push-pull tube from trim tab actuator arm.
 (4) Loosen potentiometer arm screw at potentiometer shaft and remove arm from shaft.
 (5) Locate actuator electrical plug in tailcone, disconnect plug, and cut electrical wire (approximately 6 inches from plug) to allow for splicing later.
 (6) Tie a string (approximately 4 feet long) to cut end of electrical wires.



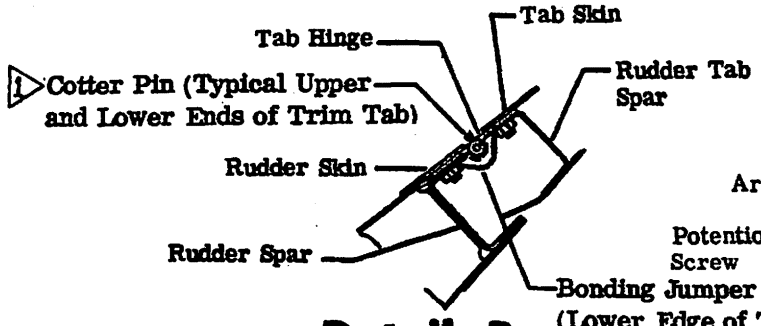


Detail A

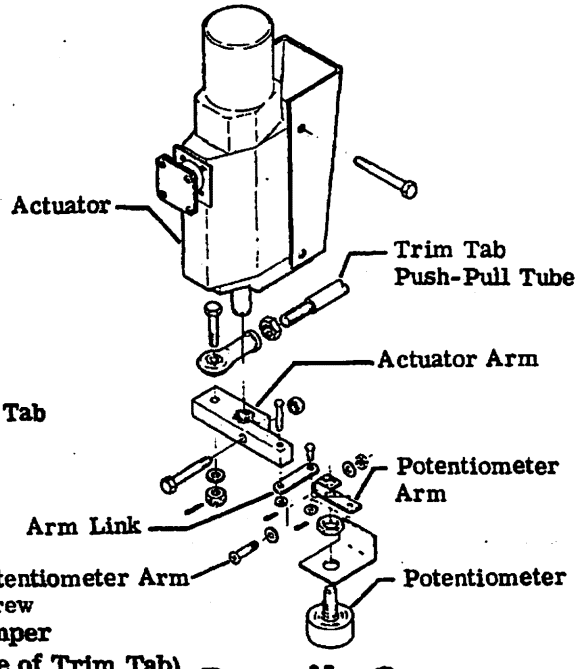


TRIM TAB CLEARANCES

1 Effective 25-167 and Subsequent and Earlier Aircraft Modified per SB23/24/25-250.



Detail B



Detail C

Rudder Trim Tab Installation
Figure 201

(7) Remove attaching parts and actuator with trim tab push-pull tube, actuator arm, link and potentiometer arm attached from the rudder, pulling electrical wiring free from rudder torque tube. Remove string from electrical wiring, leaving string in torque tube for use when reinstalling actuator.
(8) Loosen actuator arm bolt and remove actuator arm from actuator shaft.

D. Install Rudder Trim Tab Actuator (See figure 201)
(1) Align actuator arm on actuator shaft with index spot on arm aligned with index line on actuator shaft. Tighten actuator arm bolt.
(2) Install actuator with trim tab push-pull tube, actuator arm, link, and potentiometer arm attached on rudder using attaching parts.
(3) Connect push-pull tube to trim tab, align potentiometer arm on potentiometer shaft, and tighten screw.
(4) Tie string to actuator electrical wiring and pull electrical wiring through torque tube and splice electrical wiring to electrical plug wiring. Connect plug.
(5) Check rudder trim tab for proper operation. Refer to 27-20-00.
(6) Install lower access cover on rudder leading edge.
(7) Install rudder bellcrank access cover on tailcone.

E. Remove Rudder Trim Tab Actuator Potentiometer (See figure 201)
(1) Remove lower access cover from rudder leading edge.
(2) Loosen potentiometer arm screw at potentiometer shaft and remove arm from shaft.
(3) Remove potentiometer attaching nut and potentiometer from bracket.
(4) Disconnect electrical wiring at potentiometer and remove potentiometer from rudder.

F. Install Rudder Trim Tab Actuator Potentiometer (See figure 201)
(1) Connect potentiometer electrical wiring in rudder to potentiometer.
(2) Install potentiometer on bracket in rudder using existing hardware. Tape trim tab in neutral position.
(3) Using an ohmmeter, rotate potentiometer shaft to center position (±10 ohms). Position potentiometer arm on potentiometer shaft and tighten potentiometer arm screw.
(4) Check rudder trim tab for proper operation. Refer to 27-20-00.
(5) Install lower access cover on rudder leading edge.

3. Adjustment/Test

A. Rig Rudder Trim Tab System

(1) Remove lower access cover from rudder leading edge.

(2) Actuate trim tab to the streamline neutral position using the yaw trim switch. Attach a protractor to the rudder.

(3) Check trim tab travel for 15° (±1°) both left and right. Adjust limit switches on actuator to obtain proper travel.

(4) Cycle trim tab actuator several times to check for proper operation and travel. (Refer to Rudder Control System Rigging, 27-20-01, figure 202 for trim tab dimensions.)

NOTE: Check for correlation between trim tab position and rudder trim indicator. Refer to Inspection/Check.

(5) Install lower access cover on rudder leading edge.

4. Inspection/Check

A. Check Rudder Trim Tab for Proper Operation

(1) Set yaw trim switch to the RIGHT position and hold in that position until trim tab has reached its travel extreme.

NOTE: The trim tab actuator will continue to run until the switch lever is released or until the trim tab reaches a travel extreme that is controlled by internal stops of the actuator.

(2) The trim tab trailing edge will move to the left in a smooth, responsive movement.

(3) The trim indicator on the center pedestal will indicate corresponding trim tab travel.

(4) Set the yaw trim switch to the LEFT position and hold in that position until trim tab has reached its travel extreme.

(5) The trim tab trailing edge will move to the right in a smooth, responsive movement.

(6) The rudder trim indicator will indicate corresponding trim tab travel.

RUDDER PEDALS - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Rudder Pedals (See figure 201)

(1) Disconnect bob weight mechanical linkage at copilot's control column.

(2) Remove floorboards and inner skin, as required, to gain access to rudder pedal support assemblies. Insert rigging pin in rudder pedal bellcrank.

(3) Disconnect push-pull tube at torque tube arm and power brake link at brake lever.

(4) Remove attaching parts securing outboard end of rudder pedal support stud.

(5) When removing copilot's inboard pedal, disconnect rudder pedal followup linkage from pedal.

(6) Pull support stud inboard as required and remove rudder pedals and supports from the aircraft.

B. Install Rudder Pedals (See figure 201)

NOTE: If rudder pedal support stud was removed, support stud installation is accomplished from inside nose wheel well. Prior to installing support stud, assure that washer, nut and cotter pin are installed on inboard end of stud.

(1) Align inboard rudder pedal support assembly bearings with pedal support stud hole at wheel well structure and insert support stud through bearings.

(2) Align outboard rudder pedal support assembly bearings with pedal support stud hole in structure and insert support stud through bearings and structure. Secure support stud outboard end with attaching hardware.

(3) Connect push-pull tube at torque tube arm and power brake link at brake lever.

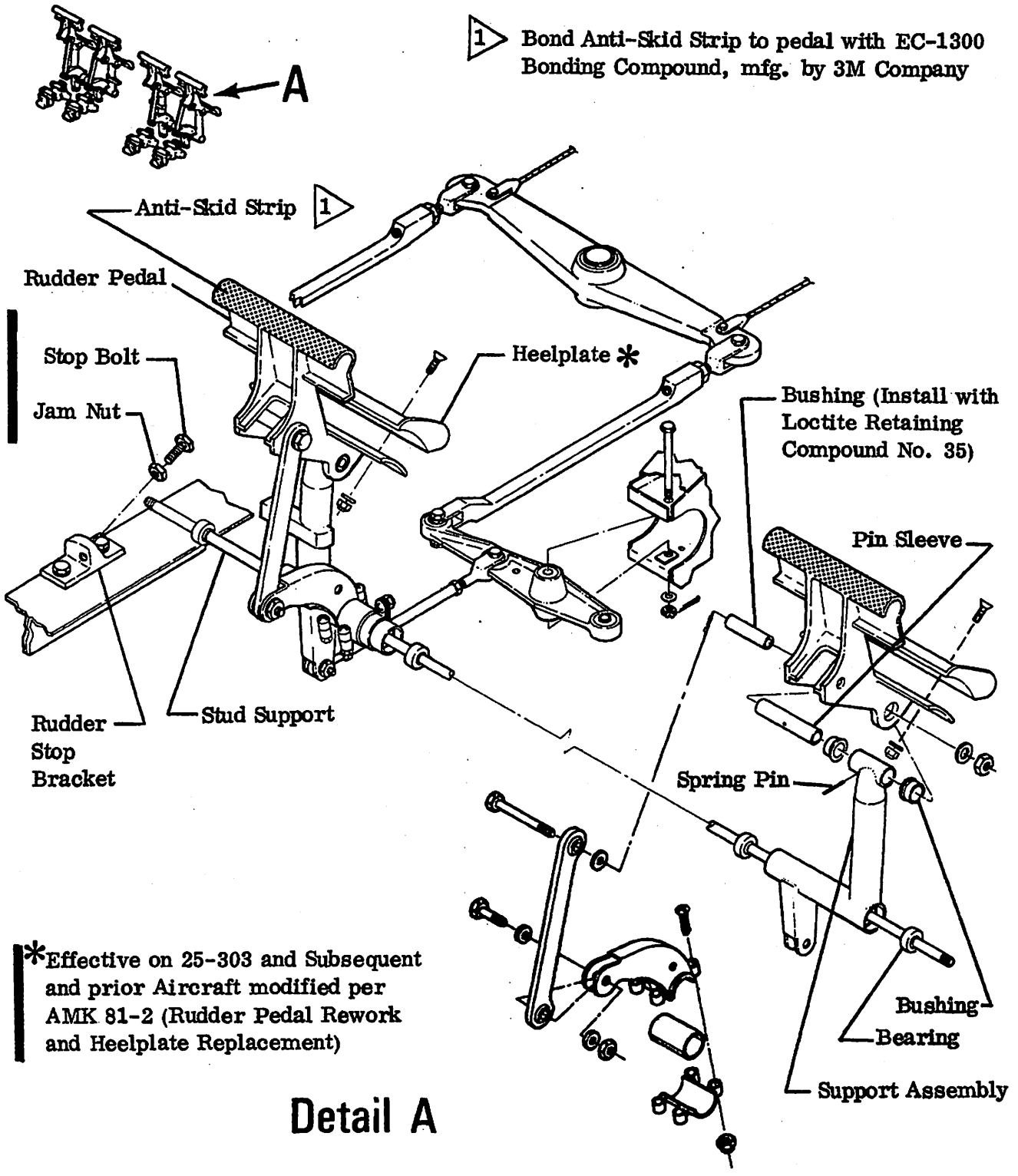
(4) Check rudder pedal rigging. Refer to 27-20-00.

(5) Rig power brakes. Refer to 32-40-00.

(6) Install and secure all floorboards and inner skins.

(7) Connect bob weight mechanical linkage to control column.

(8) Connect rudder pedal followup linkage to copilot's inboard rudder pedal.



Rudder Pedal Installation
Figure 201

RUDDER TRIM POSITION INDICATOR - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: Refer to 27-40-03 for removal and installation procedures for trim position indicator.

2. Adjustment/Test

A. Rudder Trim Position Indicator Calibration (See figure 201)

(1) The rudder trim tab potentiometer must be approximately centered with tab in neutral position prior to calibrating the rudder trim indicator.

(a) Insert small screwdriver through holes in center pedestal and, using potentiometers, calibrate position indicators as follows:

(b) Actuate rudder trim tab to neutral position.

(c) Adjust Yaw Zero potentiometer to obtain zero indication on the rudder trim indicator.

(d) Actuate rudder trim tab to full left.

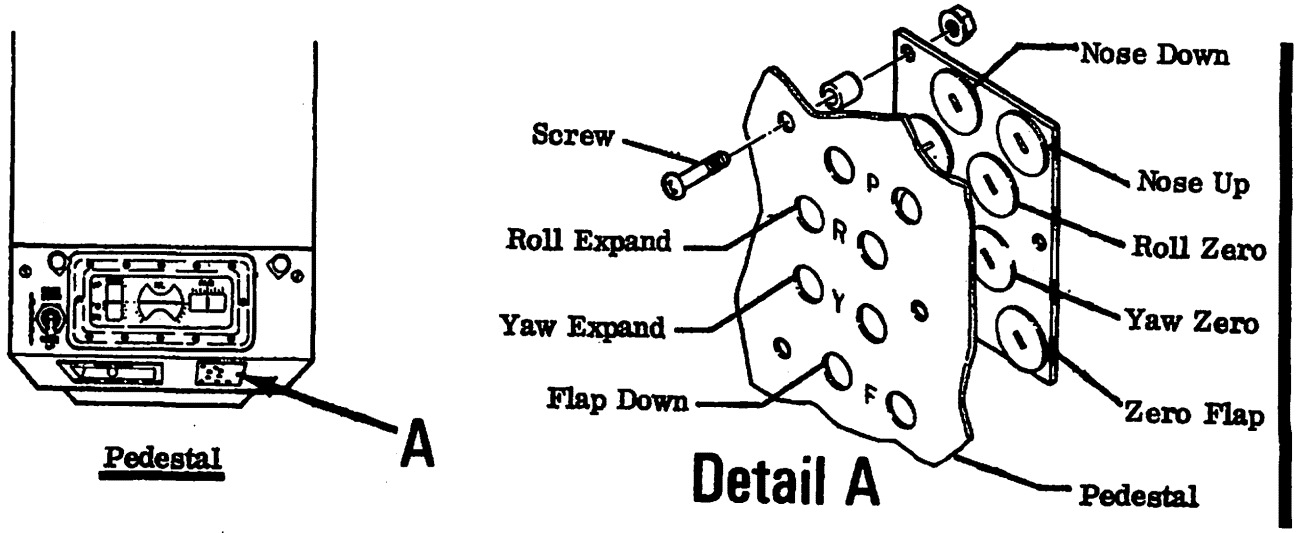
(e) Adjust Yaw Expand potentiometer to obtain full left indication on the rudder trim indicator.

(f) Actuate rudder trim tab to full right.

(g) Adjust Yaw Expand to obtain full right indication on the rudder trim indicator.

(h) Cycle rudder trim tab while checking respective indications on the rudder trim indicator.

(i) Readjust Yaw Expand potentiometer as necessary to obtain coordinated indications.



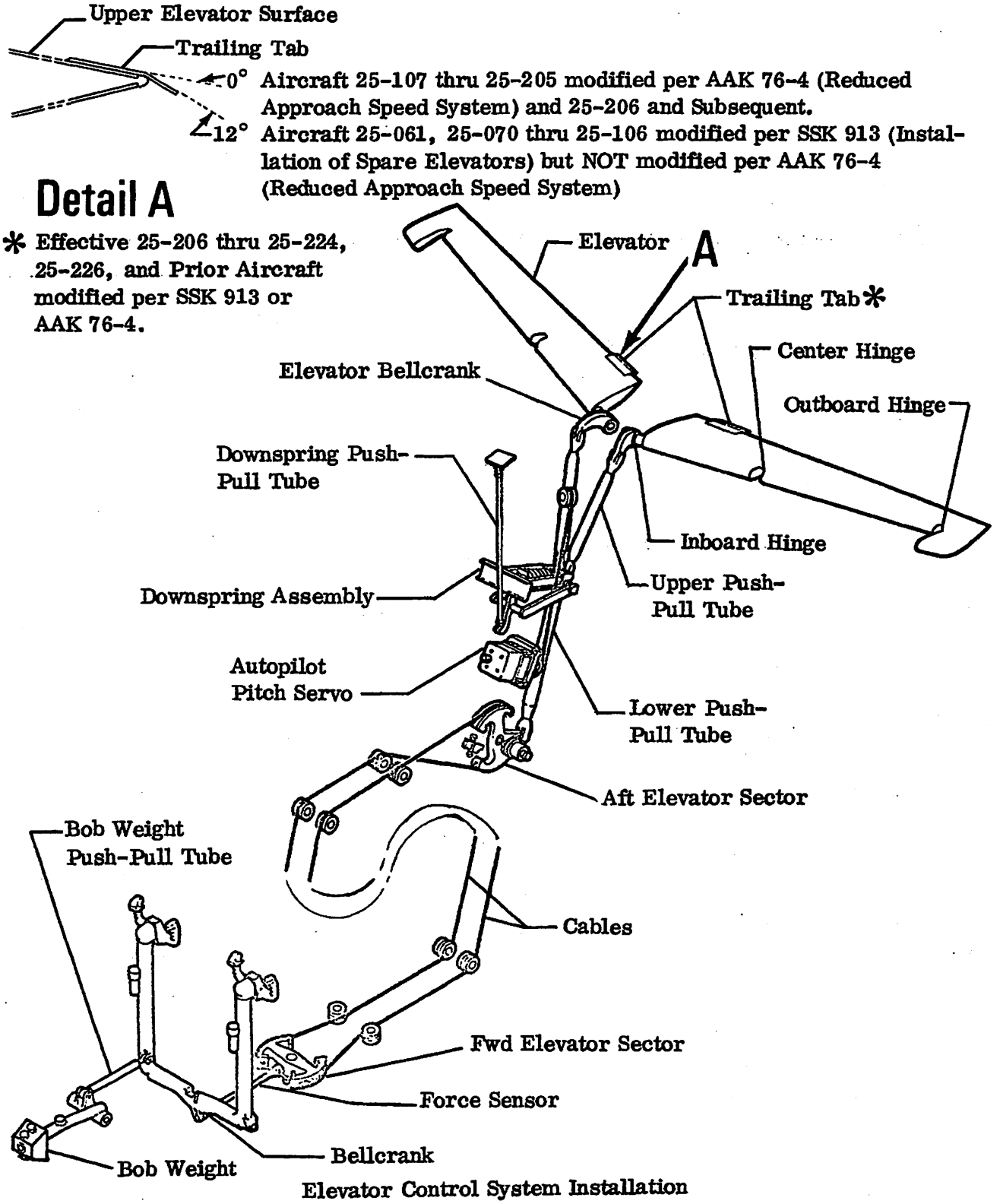
Position Indicator Calibration Assembly
Figure 201

8-107A

ELEVATOR - DESCRIPTION AND OPERATION

1. Description (See figure 1)
 A. The elevators provide longitudinal control of the aircraft. Each elevator is connected to the horizontal stabilizer at three hinge points. The mechanical control system consists of control column torque tube, bellcranks, sectors, cables, and push-pull tubes. Mounted on the control column torque tube is a bellcrank which, through a push-pull tube, provides rotary motion to the forward elevator sector. The forward elevator sector is connected to the aft elevator sector by a series of cables and pulleys. The aft sector operates the elevators through bellcranks, push-pull tubes, and an idler assembly. Access to the elevator control cables and push-pull tubes is provided by access panels in the floorboard, lower fuselage, aft tailcone, and vertical fin.
 B. For information on the force sensor, refer to Chapter 22.

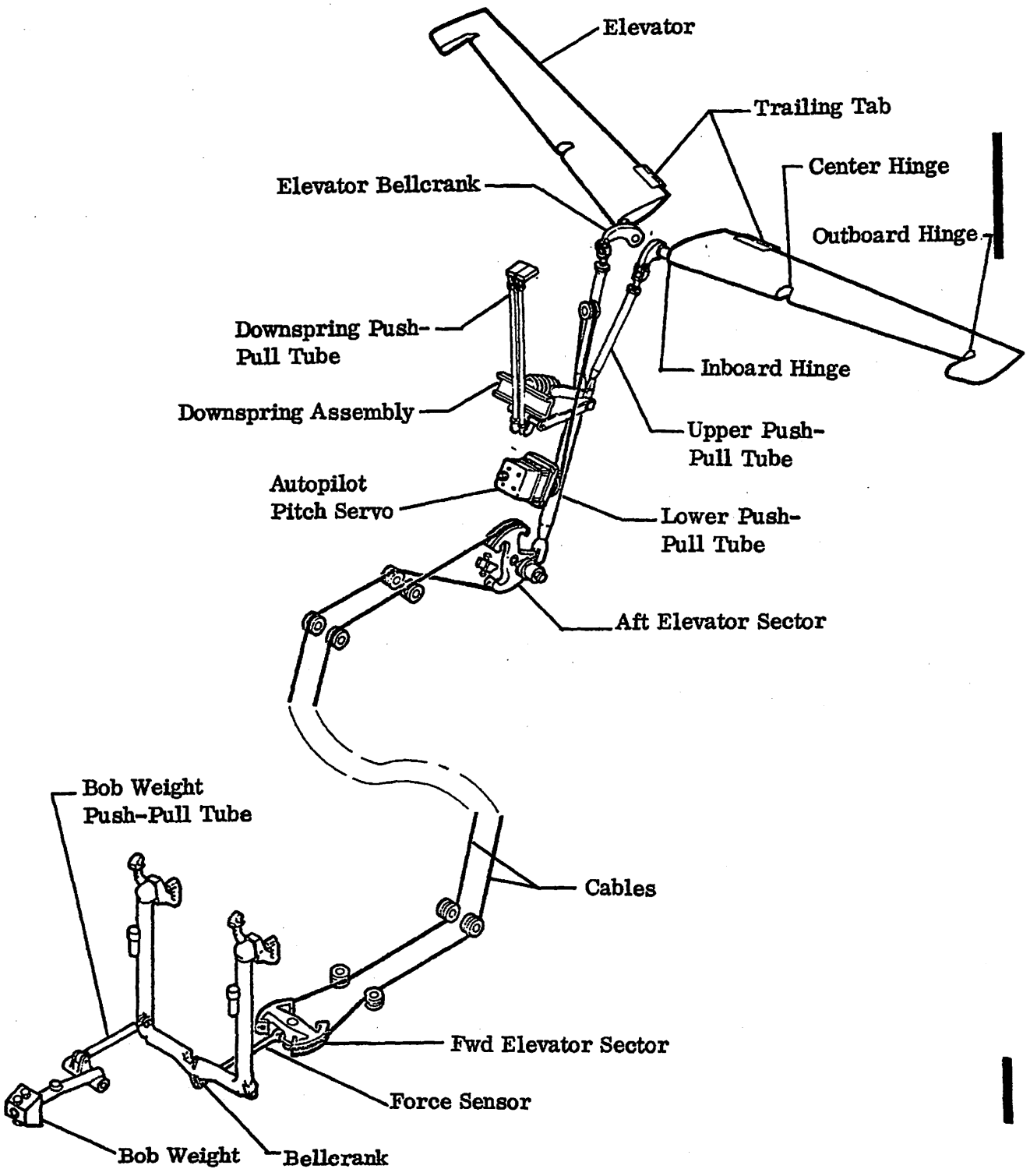
2. Operation
 A. Elevator control motion is transferred from the interconnected control columns, through a push-pull tube, bellcrank assembly, and a closed loop cable assembly to a sector assembly in the tailcone. From the sector assembly, motion is transferred through push-pull tubes to elevator horns. A bob weight, linked to the right control column, extends forward through frame 5 into the nose compartment to provide an adjustable static balance device in the elevator control system. To decrease horizontal stabilizer pitch trim force, a downspring installation is provided. The downspring interconnects the horizontal stabilizer and elevator in such a manner that movement of the stabilizer creates tension on the downspring which causes a proportional movement of the elevator in the opposite direction. The trailing tab biases the effect of the downspring. A force sensor is installed connecting the control column torque tube and the forward elevator sector to provide a monitoring function for the mach trim system. The autopilot servo is connected to the idler arm assembly by servo cables. The autopilot capstan slip clutch allows pilot override of the autopilot system.



Elevator Control System Installation
Figure 1 (Sheet 1 of 2)

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EFFECTIVITY: 25-061, 25-070 THRU 25-224 AND 25-226



Elevator Control System Installation
Figure 1 (Sheet 2 of 2)

14-42C-1

ELEVATOR - MAINTENANCE PRACTICES

1. Removal/Installation

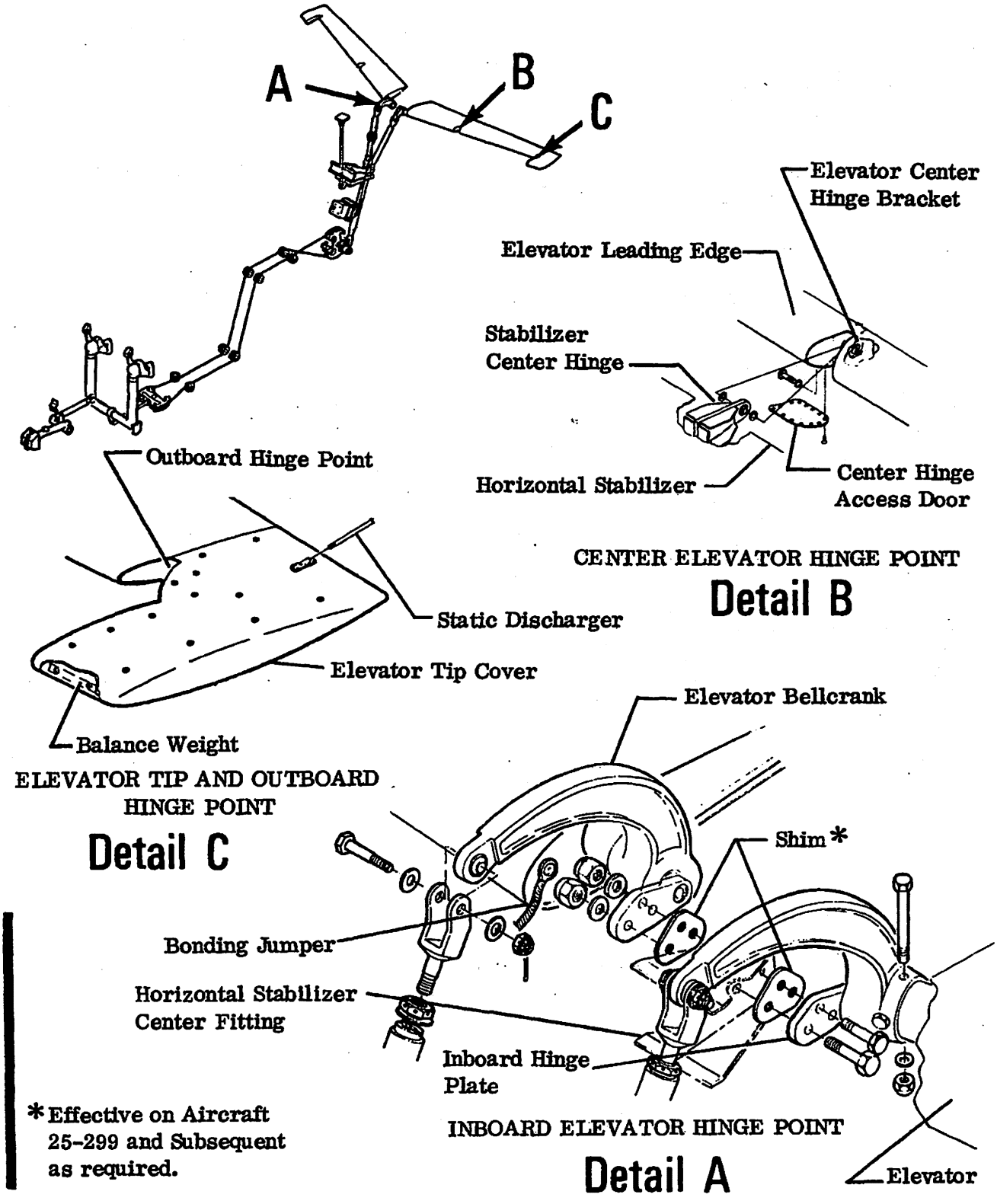
NOTE:
- Removal/Installation procedures are typical for both right and left elevators.
- Rigging pin must be inserted into aft elevator sector prior to disconnecting the upper push-pull tubes.
- When elevator tip cover is to be replaced, refer to 55-20-01.

A. Remove Elevator (See figure 201)
 (1) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.
 (2) Remove fin tip upper forward fairing from vertical stabilizer.
 (3) Remove center hinge access door and elevator tip cover.
 (4) Remove bolt connecting elevator bellcrank and upper push-pull tubes.
 (5) Remove inboard, center, and outboard hinge bolts from elevator and disconnect bonding jumper on inboard end of elevator.
 (6) Remove elevator from aircraft.

B. Install Elevator (See figure 201)
 (1) Assure that elevator balance is correct. Balance should be 10.0 to 10.5 inch-pounds overbalance. (Refer to 27-00-00.)
 (2) Connect bonding jumper to horizontal stabilizer at inboard stabilizer hinge point.
 (3) Install elevator hinge bolts securing elevator to horizontal stabilizer at the inboard, center, and outboard hinge points.

 NOTE: On some aircraft the inboard hinge bearings may be secured with Hi-Lok fasteners which are not to be reused. If any shims were removed at the inboard bearing location, they must be reinstalled.

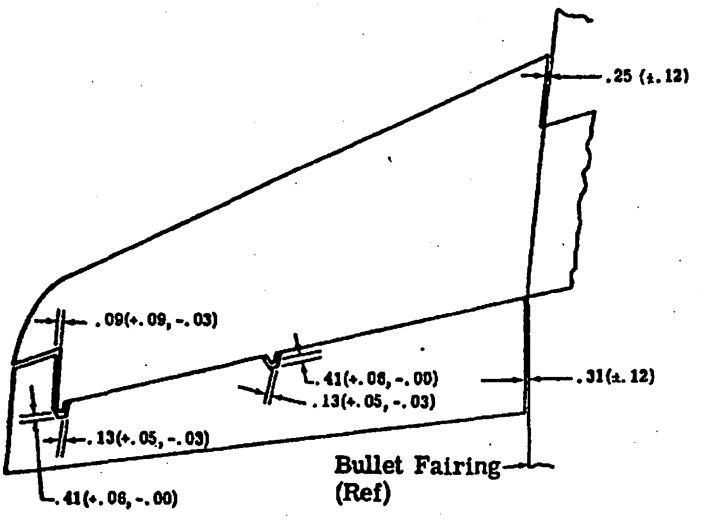
 (4) Connect elevator bellcrank to upper push-pull tube in vertical stabilizer.
 (5) Install leading edge access cover and elevator tip cover.
 (6) Check elevator for proper operation and travel. Refer to 27-30-00.
 (7) Install vertical stabilizer fin tip upper forward fairing.
 (8) Install horizontal stabilizer fairings on both sides of vertical stabilizer.



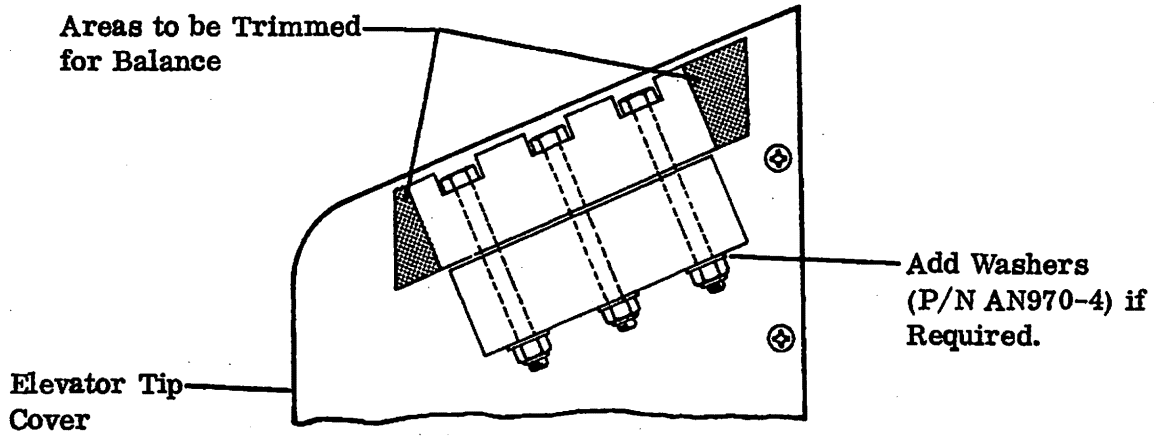
*Effective on Aircraft 25-299 and Subsequent as required.

Elevator Installation
Figure 201 (Sheet 1 of 2)

14-42C
AB8-14C-1



ELEVATOR CLEARANCES



Elevator Balance Weight

Elevator Installation
Figure 201 (Sheet 2 of 2)

2. Adjustment/Test

A. Rig Elevator Control System (See figure 202)

(1) Remove floorboards and external access panels as required to gain access to forward and aft elevator sectors.

(2) Set forward and aft elevator sectors in neutral and insert rigging pins.

(3) Adjust force sensor between the forward elevator sector and control column bellcrank to set control column in neutral. On aircraft 25-061, 25-070 thru 25-192, neutral is 3° (±1°) aft of vertical. On aircraft 25-193 and Subsequent, neutral is 5°(±30') aft of vertical.

NOTE: An alternate method of obtaining correct control column angle is to measure the angle between the column and the floorboard. The floorboard must be secured to obtain this angle. On aircraft 25-061, 25-070 thru 25-192, the angle between the column and the floorboard shall be 96° (±1°). On aircraft 25-193 and subsequent, the angle between the column and the floorboard shall be 94° (±30').

(4) Position horizontal stabilizer to neutral (0.5° stabilizer nose down).

NOTE: The following instructions are applicable only with horizontal stabilizer set to neutral (0.5° stabilizer nose down) position since elevator travel varies with stabilizer setting.

(5) Assure that elevators are properly synchronized and position contour plate on horizontal stabilizer and elevators. Adjust the push-pull tubes to set elevators in neutral with respect to the horizontal stabilizer.

NOTE: The upper push-pull tubes are used to synchronize the elevators with each other while the lower push-pull tube adjusts for elevator neutral position.

(6) Remove access panel from left side of vertical stabilizer gaining access to the downspring installation.

(7) Remove rigging pins from forward and aft elevator sectors and position horizontal stabilizer to 3°44' down using the rigging plate.

(8) Move elevators until downspring caging pin hole is exposed.

WARNING: THE EXISTING SPRING TENSION OF THE CAGED SPRING ASSEMBLY IS APPROXIMATELY 237 POUNDS. USE EXTREME CAUTION WHEN REMOVING ATTACHING PARTS AND SPRING ASSEMBLY FROM AIRCRAFT.

(9) Insert caging pin and remove bolt and nut attaching drawbolt to frame assembly.

(10) Fabricate rigging pin from 3/16-inch diameter steel rod approximately 2.25 inches long.
(11) Insert rigging pin into frame assembly as shown in figure 202.
(12) Loosen jammut and adjust downspring push-pull tube(s) until rigging pin can be inserted into the right and left hinge brackets as shown.
(13) Tighten jammut on push-pull tube. Remove rigging pin from frame assembly.
(14) Attach drawbolt to frame assembly using bolt and nut. Remove caging pin.
(15) Position horizontal stabilizer to neutral position. Adjust pitch servo cable tension to 60 (±5) pounds at 75°F. Adjust elevator control tension to 75 (±5) pounds at 75°F. (See tension temperature-load corrections, 27-00-00.)
(16) Safety turnbuckles.
(17) Attach clinometer to elevator to measure travel.
(18) Set the primary up stop (located at aft sector of tailcone) to obtain elevator T.E. up travel of 15° (+1°/-0°). Secure upstop bolt with jammut. On Aircraft 25-198 and Subsequent and previous aircraft incorporating SB 23/24/25-266, safety wire stop bolt.
(19) Set the primary down stop (located at aft sector in tailcone) to obtain elevator T.E. down travel of 15° (+1°/-0°). Secure down stop bolt with jammut. On aircraft 25-198 and Subsequent and previous aircraft incorporating SB 23/24/25-266, safety wire stop bolt.
(20) Adjust the secondary stop bolts on the forward elevator sector to contact their stops when 100 (±25) pounds of force is applied to the control column wheel (fore and aft) after the primary stop bolts are contacted. Secure secondary stop bolts with jam nuts. On aircraft 25-198 and Subsequent and previous aircraft incorporating SB 23/24/25-266, safety wire stop bolts.
(21) With control column in neutral, adjust bob weight push-pull tube so bob weight clears aircraft structure equally in both the up and down positions.
(22) Move control column forward and aft until secondary stops are contacted. Note position of bob weight with respect to aircraft structure at each extreme. Readjust push-pull tube as required.
(23) Adjust bob weight stops so the secondary stops and bob weight stops are contacted simultaneously in either the up or down position.
(24) Operate the elevators several times while checking for freedom of movement.
(25) Install and secure floorboards and access panels.

3. Inspection/Check

A. Operational Check of Elevator System

(1) During operational check, observe response and freedom of movement of control columns and elevators.

(2) Pilot and copilot control columns are interconnected and column movements should be synchronized.

(3) Push pilot control column forward from neutral.

(4) Observe the movement of the elevators in a downward direction.

(5) Relax force applied to the control column and observe elevator movement back to the neutral position.

(6) Pull control column aft from neutral.

(7) Observe the movement of the elevators in an upward direction.

(8) Relax force applied to the control column and observe elevator movement back to the neutral position.

B. Elevator Control System Friction Check

(1) Position horizontal stabilizer full nose up (aircraft nose down).

(2) Place a push-pull spring scale (accurate within 1/2 pound) on the pilot's control wheel hand grip so that the force applied is through the horizontal centerline of the control wheel hub.

(3) Slowly pull horizontally (aft) and read the force applied to move the control column off of the forward stop. The maximum allowable force is 27 pounds.

(4) Continue pulling aft and read the force applied to move the column through neutral. The maximum allowable force is 29 pounds.

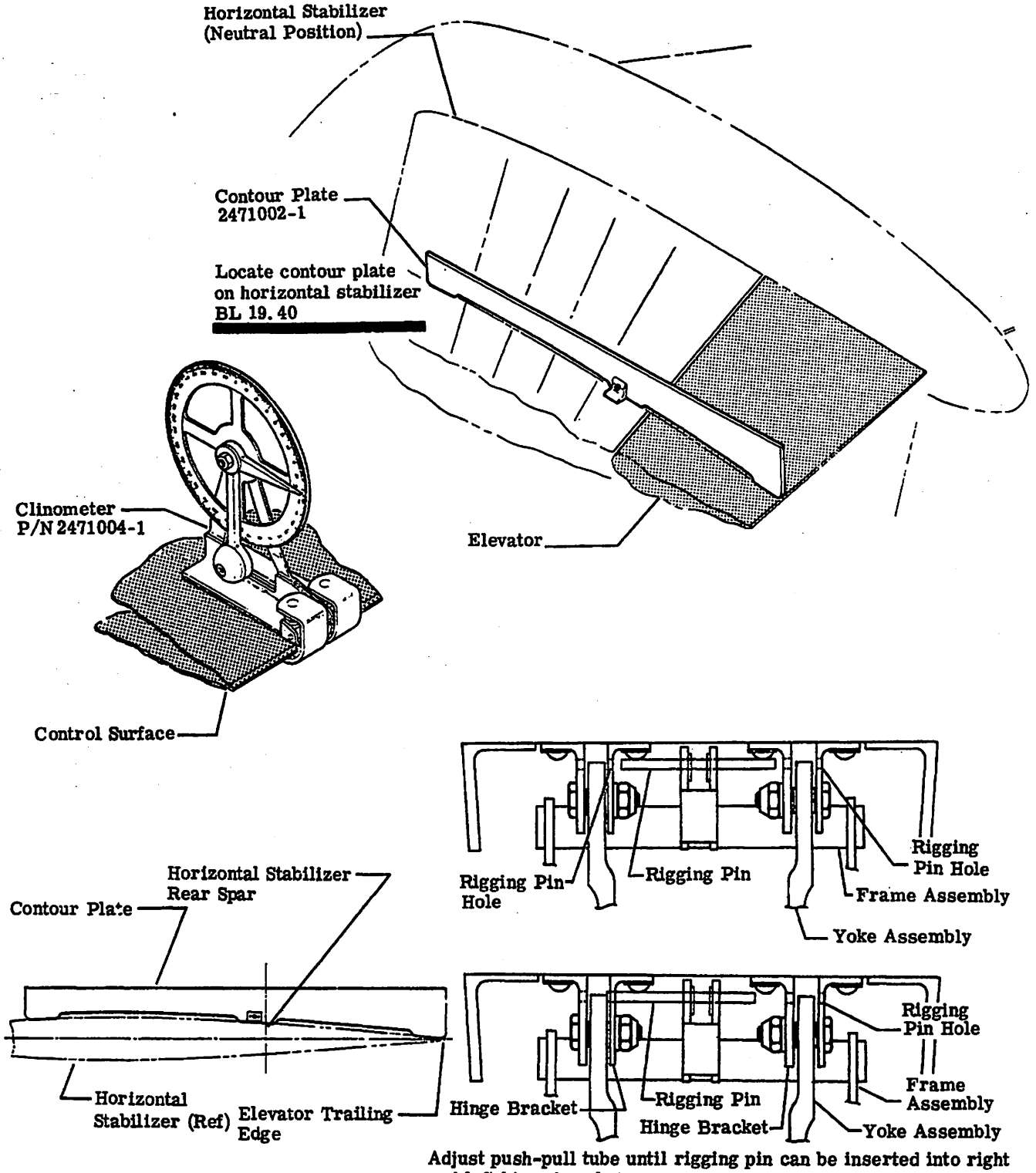
(5) Position horizontal stabilizer full nose down (aircraft nose up) and the control column well aft of neutral.

(6) Relax the spring scale tension to allow the control column to move slowly forward and read the force as the control column moves through neutral. The minimum allowable force is 3 pounds.

(7) Continue to relax tension on the spring scale slowly and read the force just before the control column reaches full forward. The minimum allowable force is one pound.

(8) If pull force is greater or relax force is less than that specified for any point, check for proper cable tensions; check pulleys, sectors, and bellcranks for freedom of movement; inspect system for binding; check system for proper lubrication; and ensure that elevator control system is properly rigged.

(9) After all discrepancies have been corrected, perform steps (1) through (7) and ensure that the forces are as specified.



Adjust push-pull tube until rigging pin can be inserted into right and left hinge brackets.

Elevator Control System Rigging
Figure 202

ELEVATOR DOWNSPRING - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Elevator Downspring (See figure 201)

(1) Remove access panels from vertical stabilizer to gain access to elevator downspring installation.

(2) Move elevators to either the full up or down position.

(3) Insert caging pin through drawbolt (fabricate caging pin from steel rod 3/16 inch in diameter at 3.00 inches in length).

WARNING: THE EXISTING SPRING TENSION OF THE CAGED SPRING ASSEMBLY IS APPROXIMATELY 237 POUNDS. USE EXTREME CARE WHEN REMOVING ATTACHING PARTS AND SPRING ASSEMBLY FROM AIRCRAFT.

(4) Remove attaching parts securing forward end of spring assembly to frame assembly.

(5) Loosen tension on pitch servo cables.

(6) Remove attaching parts securing pitch servo cables and aft end of spring assembly to yoke assembly. Remove spring assembly from aircraft.

B. Install Elevator Downspring (See figure 201)

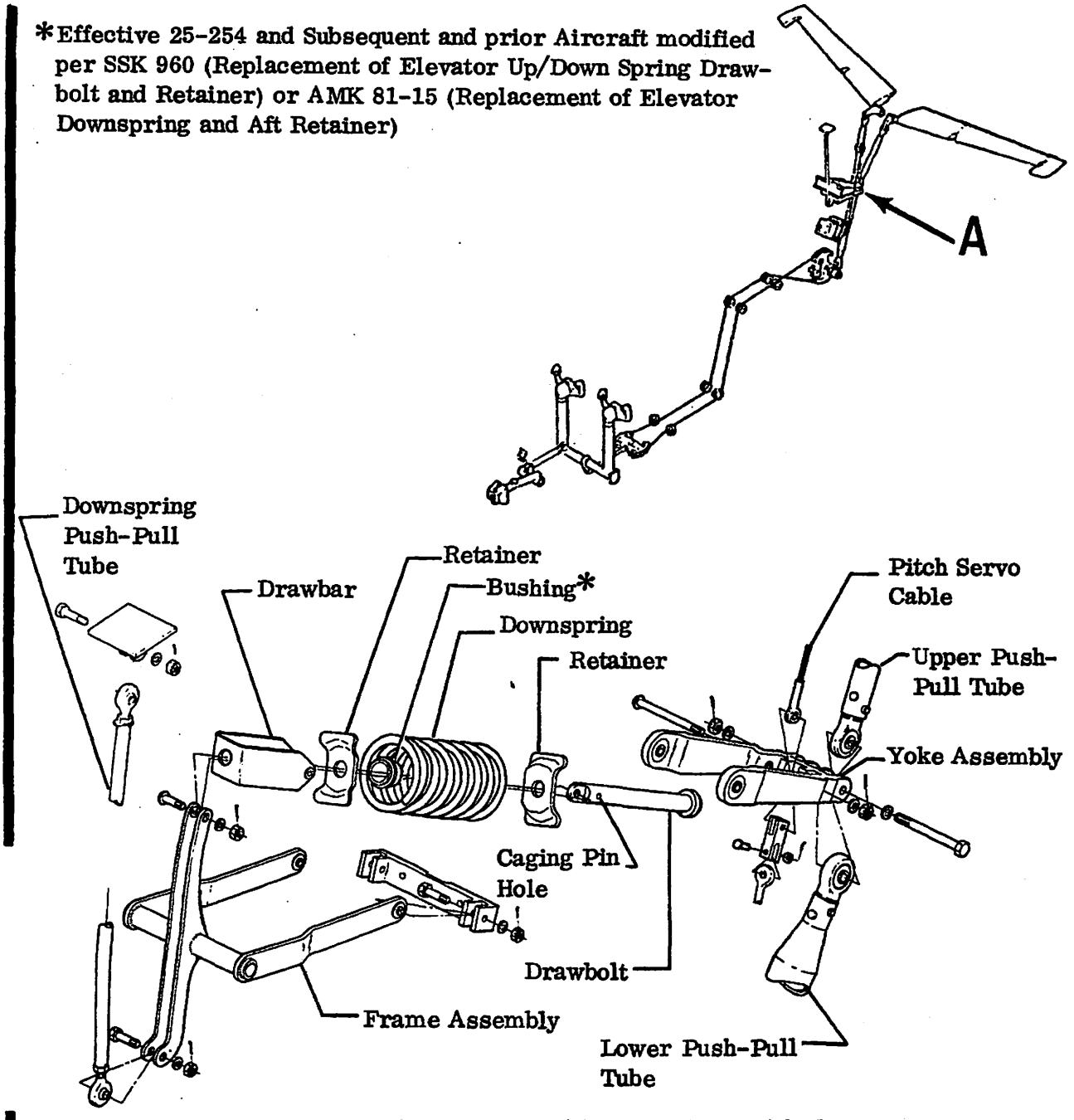
(1) Install spring assembly and pitch servo cables and secure with attaching parts.

(2) Secure forward end of spring assembly to frame assembly with attaching parts.

(3) Set elevator to either full up or down and remove caging pin.

(4) Install access panels on horizontal stabilizer.

*Effective 25-254 and Subsequent and prior Aircraft modified per SSK 960 (Replacement of Elevator Up/Down Spring Drawbolt and Retainer) or AMK 81-15 (Replacement of Elevator Downspring and Aft Retainer)



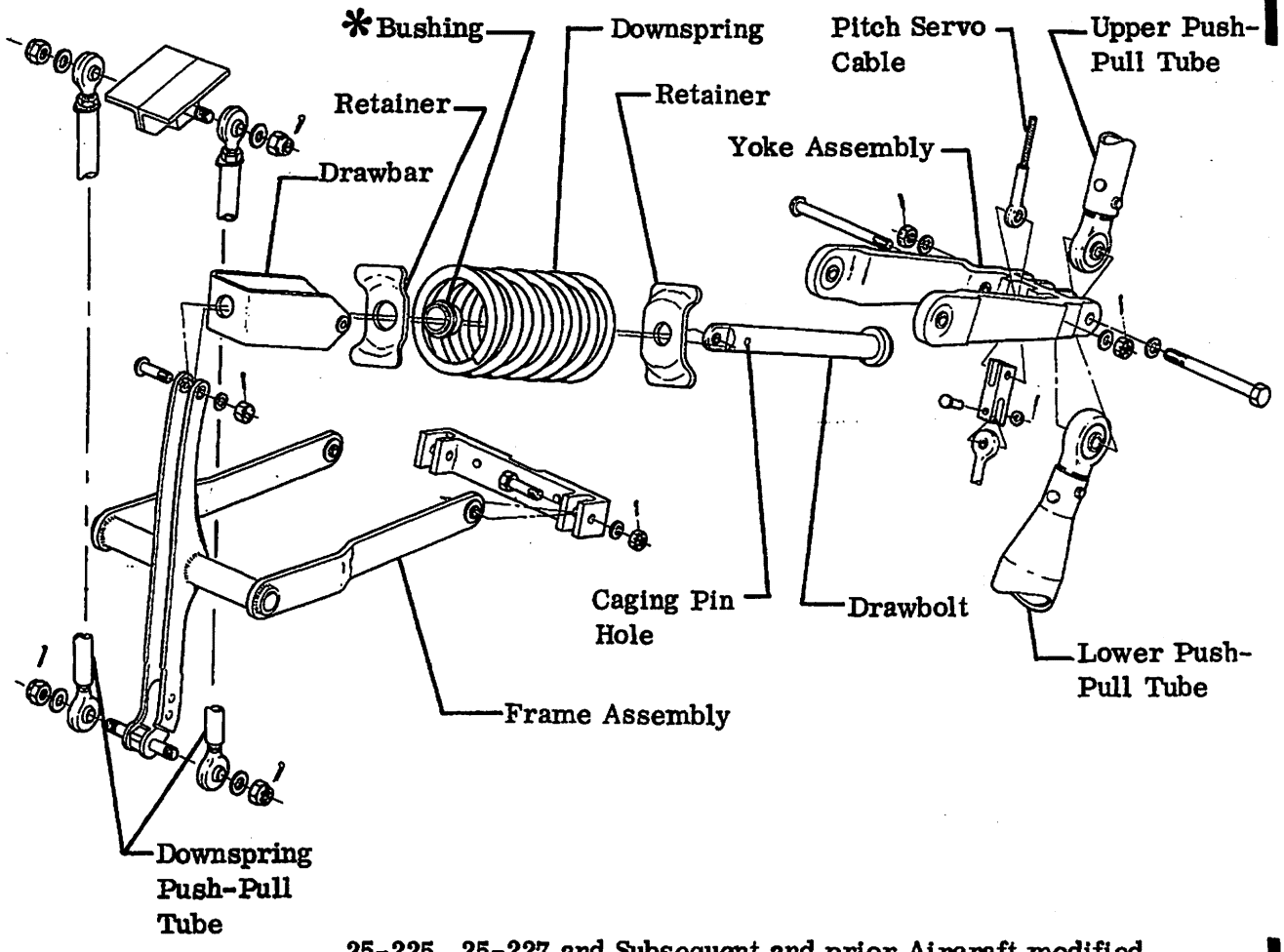
25-061, 25-070 thru 25-224 and 25-226 NOT modified per AAK 76-4 (Reduced Approach Speed System) or AMK 77-9A (Installation of Dual Elevator Up/Down Spring Secondary Retention Pushrods)

Detail A

Elevator Downspring Installation
Figure 201 (Sheet 1 of 2)

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* Effective 25-254 and Subsequent and prior Aircraft modified per SSK 960 (Replacement of Elevator Up/Down Spring Drawbolt and Retainer) or AMK 81-15 (Replacement of Elevator Downspring and Aft Retainer)



25-225, 25-227 and Subsequent and prior Aircraft modified per AAK 76-4 (Reduced Approach Speed System) or AMK 77-9A (Installation of Dual Elevator Up/Down Spring Secondary Retention Pushrods)

Detail A

Elevator Downspring Installation
Figure 201 (Sheet 2 of 2)

ELEVATOR BOB WEIGHT - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Bob Weight (See figure 201)

(1) Gain access to the bob weight push-pull tube. Disconnect pugh-pull tube from control column and bob weight bellcrank assembly.

(2) Remove nose compartment access door.

(3) Remove attaching parts and bob weight cover.

(4) Remove attaching parts and remove weight from bellcrank assembly.

(5) Remove attaching parts and remove bellcrank box assembly.

B. Install Bob Weight (See figure 201)

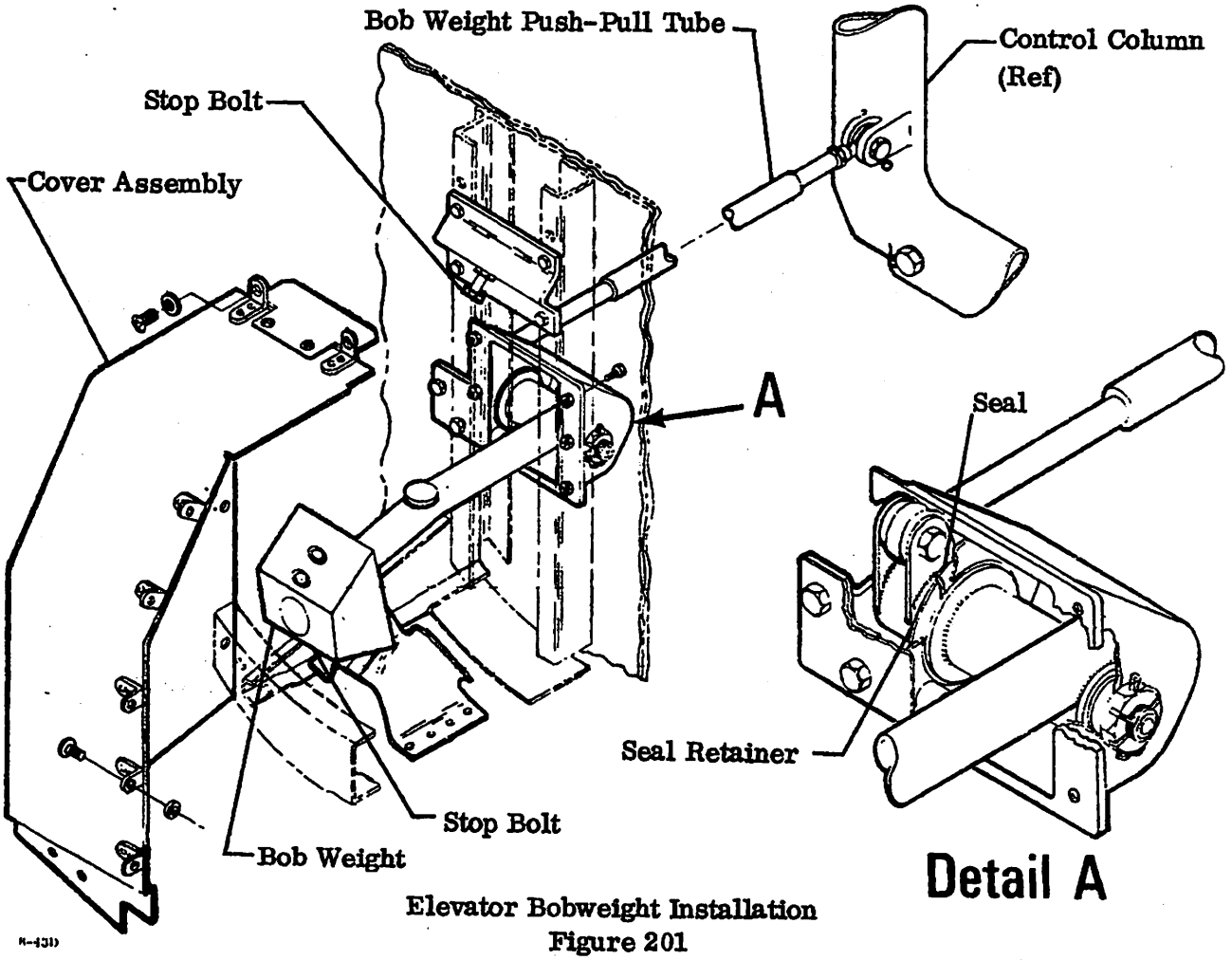
(1) Apply a faying surface seal to area where bellcrank box assembly is installed.

(2) Install attaching parts and secure bellcrank box assembly.

(3) Install weight on bellcrank assembly and secure with attaching parts.

(4) Install cover and secure with attaching parts.

(5) Install push-pull tube and secure with attaching parts.



Elevator Bobweight Installation
Figure 201

STALL WARNING - DESCRIPTION AND OPERATION

1. DESCRIPTION

NOTE: ° Due to the possible effects on stall characteristic, whenever maintenance as noted in items (1) through (4) below is performed, the aircraft must be flight tested to verify aerodynamic acceptability and stall speeds. This flight test must be conducted by a pilot approved by Gates Learjet Corporation for stall test flights. The stall warning system must be checked in accordance with the flight or ground calibration procedures in 27-31-00 of this maintenance manual when maintenance as noted in items (1) through (5) is performed.

(1) When maintenance requiring removal and installation, repair, or installation of a new wing leading edge is performed. This includes loosening and retightening any of the leading edge screws.

(2) When a tip tank strake is loosened or removed from the tip tank or a new strake is installed. Applies to 25-206 and Subsequent and prior aircraft equipped with AAK 76-4, Reduced Approach Speed System.

(3) When a new tip tank is installed. Applies to 25-206 and Subsequent and prior aircraft equipped with AAK 76-4, Reduced Approach Speed System.

(4) When maintenance requiring repair to, or installation of, new stall strips is performed. Applies to 25-290 and Subsequent and prior aircraft equipped with Stall Fence, AAK 79-10 or AMK 83-5.

(5) When a stall warning transducer, computer, or indicator is repaired, replaced, or calibrated.

° On Aircraft 25-206 and Subsequent and prior Aircraft equipped with AAK 76-4, Reduced Approach Speed System, replacement of the stall transducer vane is permitted in the event of a failed heater. Physical damage to the vane requires total transducer replacement because of possible damage to the transducer shaft, bearing, and potentiometer. Vane replacement requires functional test to verify that shaker and pusher actuation occurs within the prescribed tolerance of the vane plate marks.

° If angle-of-attack indicator replacement does not require an IND or R_X potentiometer adjustment and a quick check shows indicator to be at intersection of yellow/green for shaker and at intersection of red/yellow for pusher then the aircraft does not require flight test/adjustment or ground calibration adjustment.

° Perform functional test of stall warning system in accordance with the current inspection interval specified in Chapter 5.

NOTE: ° A flashing stall warning annunciator indicates proper system operation. A steady stall warning annunciator indicates system malfunction unless stall switches are off or angle of attack needle is in red area.

° Angle-of-attack vanes must be moved slowly to give correct indication. The rate of vane movement is used in stall computation. Failure to move vanes slowly will cause angle-of-attack indicator needle movement to be jumpy.

° Components of stall warning system that do not meet functional test requirements shall be replaced or recalibrated.

A. For sake of discussion, various aircraft serial/modification groups are defined as follows:
 (1) Aircraft not equipped with 8° Flap System - Aircraft 25-061 and 25-070 thru 25-196 and 25-198 not modified per AAK 75-5, "Installation of Components for Partial Flap Extension during Takeoff - 8° Flap System" and not modified per AAK 76-4, "Reduced Approach Speed System."
 (2) Aircraft equipped with 8° Flap System - Aircraft 25-197 and 25-199 thru 25-205 and earlier aircraft modified per AAK 75-5, "Installation of Components for Partial Flap Extension during Takeoff."
 (3) Aircraft equipped with RAS - Aircraft 25-206 and Subsequent and earlier aircraft modified by AAK 76-4, "Reduced Approach Speed System."
 (4) Aircraft equipped with Softflite - Aircraft 25-290 and Subsequent and earlier aircraft modified by AAK 79-10 or AMK 83-5, "Install of Wing Fences, Stall Strips, and Boundary Layer Energizers."
 (5) Aircraft equipped with Horizontal Stabilizer Trim and Autopilot Improvement - Aircraft 25-337, 25-342 and Subsequent and prior aircraft modified per AMK 81-9 or AMK 81-7, "Horizontal Stabilizer Trim and Autopilot Improvement."
 (6) Aircraft equipped with Handling Qualities Improvement - Aircraft serials 25-067, 25-070 thru 25-205 equipped with AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."

B. The stall warning system provides the crew with an indication of an impending stall and also commands an aircraft nose down attitude as corrective action.

C. Aircraft 25-061, 25-070 thru 25-205 not modified per AAK 76-4, "Reduced Approach Speed System".
 (1) System components are two vane-actuated angle-of-attack transducers on the forward fuselage, a stall warning computer-amplifier, a stall warning bias box in the nose section, a control column shaker on each control column, a normal accelerometer attached to frame 2 in the nose section, an angle-of-attack indicator, two ON-OFF control switches, a Stall Warning Test Switch, a stall warning accelerometer cutout box, and two power-on stall warning lights located on the glareshield. The system also utilizes the elevator pitch servo.
 (2) On Aircraft with 8° flap system not equipped with AAK 76-4, "Reduced Approach Speed System," the stall warning test switch is not utilized.

D. Aircraft 25-206 thru 25-289 and prior aircraft modified per AAK 76-4, "Reduced Approach Speed System."
 (1) System components consist of two vane-actuated angle-of-attack transducers on the forward fuselage, a stall warning computer in the nose compartment, a control column shaker on each control column, a normal accelerometer in the nose compartment, two angle-of-attack indicators in the instrument panel, two ON-OFF switches, a test switch, two altitude switches, and two stall warning annunciators located on the glareshield.
 (2) The system also utilizes the elevator pitch servo.

E. Aircraft certified for operation in Australia.
 (1) System components are two vane-actuated angle-of-attack transducers on the forward fusealge, a stall warning computer-amplifier, a stall warning bias box in the nose section, a control column shaker on each control column, a normal accelereometer attached to frame 2 in the nose section, two angle-of-attack indicators, two ON-OFF control switches, a Stall Warning Test Switch, a stall warning series pusher control box, and two power-on warning lights located on the glareshield.
 (2) The system also utilzies the elevator pitch servo and aural warning horn.

F. Aircraft 25-290 and Subsequent and prior Aircraft modified per AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers."
 (1) System components consist of left and right angle-of-attack transducers on the forward fuselage, a stall warning computer and an accelerometer in the nose avionics compartment, a control column shaker on each column, and electrical circuitry acting on the autopilot pitch servo to produce control column-elevator pitch down. System control is via left and right Stall Warning ON-OFF Switches. A single Left-Right Test Switch facilitates system functional test. Visual indications are provided by stall warning annunciators and angle-of-attack indicators. High-altitude operation with it attendant higher stall speeds are programmed into the system by two altitude sensing switches in the nose avionics compartment.

G. Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified per AMK 81-7 or AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement," system components are the same as those described in paragraph 1.C. through 1.F., as applicable except that the electrical circuitry acting on the autopilot pitch servo produces a control column nudging in the shaker range.

2. OPERATION

A. Aircraft 25-061 and 25-070 thru 25-205 not modified per AAK 76-4, "Reduced Approach Speed System" including aircraft equipped with 8° Flap System. (See figures 1 and 2.)

(1) The angle-of-attack transducers are supplied with a regulated voltage which is modified by the stall warning bias box. The wiper of the angle-of-attack transducer supplies a voltage to the computer which is proportional to the aircraft angle-of-attack. When the aircraft angle-of-attack increases such that speed is 7% above aircraft stall speed, the stall warning computer energizes the control column shaker motors. The shaker motors then produce a low-frequency, high-amplitude buffet signal to the crew members. When both angle-of-attack transducer vanes simultaneously increase to 5% above a stall condition, the computer energizes the down clutch of the pitch servo to command an aircraft nose-down attitude. The amount of force applied in the nose-down direction is 80 pounds at the control wheel. The stall warning accelerometer cutout completes a holding circuit and prevents any decrease in the nose-down force by the normal accelerometer. As soon as the angle-of-atack transducer vanes decrease below the stall point, the nose-down command is removed. In the event one angle-of-attack transducer vane increases 5% above a stall condition, due to wind gust or bird strike, the shaker will energize and the computer will command a nose-down attitude. The amount of force applied is 50 pounds. This force diminishes linearly through the normal accelerometer to maintain 1/2 g downward push at the control wheel. The angle-of-attack indicator provides continuous readout of the aircraft angle-of-attack. Normal autopilot information is first fed into the stall warning computer and then to the elevator pitch servo. When the angle-of-attack transducer vanes increase to the pusher actuation point, the autopilot information is disconnected from the pitch servo. The stall warning system is powered by the battery bus.

B. Aircraft certified for operation in Australia. (See figure 3.)

(1) The angle-of-attack transducers are supplied with a regulated voltage which is modified by the stall warning bias box. The wiper of the angle-of-attack transducer supplies a voltage to the computer which is proportional to the aircraft angle-of-attack. When the aircraft angle-of-attack increases such that speed is 7% above aircraft stall speed, the stall warning computer energizes the control column shaker motors and the aural warning. The shaker motors then produce a low-frequency, high amplitude buffet signal to the crew members. If either angle-of-attack transducer vane increases to 5% above a stall,

and both shakers are activated, the computer energizes the down clutch of the pitch servo to command an aircraft nose-down atttitude and the stall warning light will be flashing. The amount of force applied in the nose down direction is 50 pounds at the control wheel. This force diminishes linearly through the normal accelerometer to maintain 1/2 g downward push at the control wheel. As sson as one angle-of-attack transducer vane or both decrease below the stall point, the nose-down command is removed. In the event one angle-of-attack transducer vane increases 5% above a stall condition, due to wind gust or bird strike, the shaker will energize and the aural warning will sound and the lights will flash. The angle-of-attack indicators provide continuous readout of the aircraft angle of attack. Normal autopilot information if first fed into the autopilot electrical box and then to the elevator pitch servo. When the angle-of-attack transducer vanes increase to the pusher actuation point, the autopilot information is disconnected from the pitch servo. The stall warning system is powered by the battery bus.

C. Aircraft 25-206 thru 25-289 and prior Aircraft equipped with AAK 76-4, "Reduced Approach Speed System." (See figure 4.)

(1) The angle-of-attack transducers are supplied with a constant voltage. As the angle-of-attack increases, the transducer output voltage decreases. The transducer signal is fed into the stall warning computer along with the flap position and altitude information. Flap position information is fed to the computer from three switches at each flap sector. The three switches provide the computer with four different flap positions; full up, 0° to 13°, 13° to 25°, and 25° to 40°. Altitude information is fed to the computer from the altitude switches which close at 22,500 (±750) feet. When the altitude switch closes, the shaker and pusher actuation speeds are increased approximately 10 knots. The computer sums all of these signals and gives an instantaneous measure of the aircraft's proximity to a stall. When the aircraft angle-of-attack increases such that speed is 7% above aircraft stall (pusher) speed, the computer energizes the control column shakers. Pusher speed must be at least 3 knots above aerodynamic stall speed. The shakers produce a low-frequency, high amplitude buffet signal to the crew members. At the same time the shakers actuate, the angle-of-attack indicator needles enter the yellow arc and the stall warning lights will illuminate and flash. If the impending stall condition continues until speed reaches approximately one knot (seven knots with flaps up) above aircraft stall (pusher) speed, the angle-of-attack indicator will reach the red line at which point the computer will command a nose-down attitude to the pitch servo. The amount of force applied in the nose-down direction is 80 pounds at the control wheel. The force is applied to the control column in a pumping motion until the angle-of-attack decreases. The accelerometer limits the pusher force to approximately 1/2 g force at the control wheel. A Stall Warning Test Switch is located on the pedestal pitch panel. The test circuit will operate only with the squat switches in the ground mode. Holding the Test Switch to RH TEST

simulates an increasing angle-of-attack which causes the RH angle-of-attack indicator needle to begin a slow sweep across the green band. As the needle crosses into the yellow band, the RH control column shaker will begin to operate and the R STALL warning light will begin to flash. The needle will continue to the red line. As the needle reaches the red line, the pusher will actuate. Testing the LH stall warning is the same.

D. Aircraft 25-290 thru 25-336 and 25-338 thru 25-341 and prior Aircraft modified per AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers." (See figure 4.)

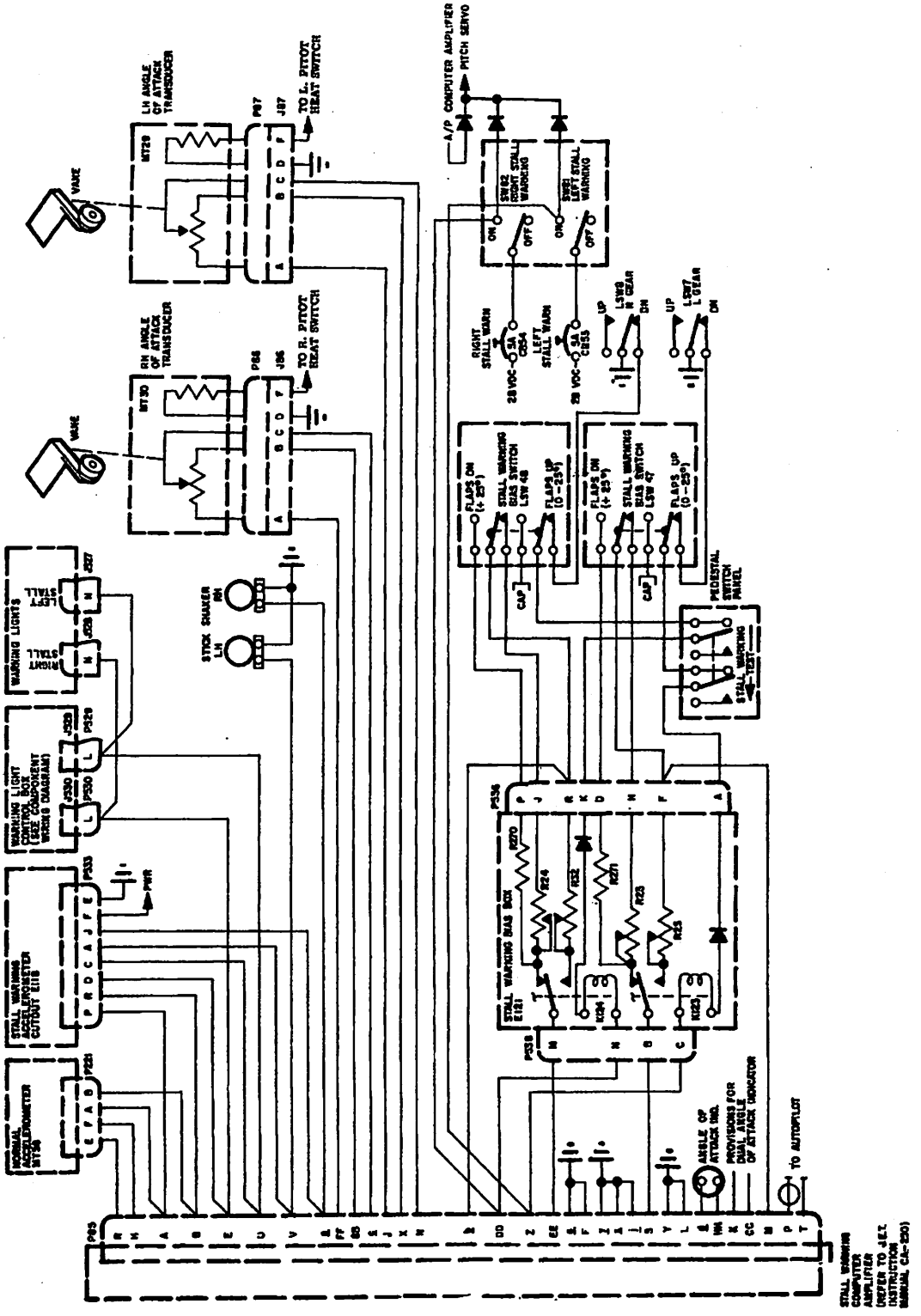
(1) The angle-of-attack transducers are supplied with a constant voltage. As the angle-of-attack increases, the transducer output voltage decreases. The transducer signal is fed into the stall warning computer along with the flap position and altitude information. Flap position information is fed to the computer from three switches at each flap sector. The three switches provide the computer with four different flap positions; full up, 0° to 13°, 13° to 25°, and 25° to 40°. Altitude information is fed to the computer from the altitude switches which close at 22,500 (±750) feet. When the altitude switch closes, the shaker and pusher actuation speeds are increased approximately 10 knots. The computer sums all of these signals and gives an instantaneous measure of the aircraft's proximity to a stall. If airspeed is reduced and the angle of attack increased to 7% above stall speed, the computer energizes the control column shaker. The stall warning lights will flash at shaker pulse rate and the angle-of-attack indicator needle will move into the dial face yellow band. At this point the shakers produce a relatively low-frequency pulse rate. If the angle of attack is increased to within ±3 knots of stall speed (stall speed being defined as pusher actuation speed at forward c.g. in any flap configuration), the angleof-attack indicator needles will cross into the red line. At this point the computer will command the autopilot pitch servo to produce a rapid nose-down attitude, exerting an 80-pound force at the control column. Pusher pitch-down will be modified by the normal accelerometer which prevents excessive negative g forces (1/2 g maximum) from being imposed on the aircraft. Shakers will continue to operate, but at a much higher pulse rate. Since the stall warning lights are directly associated with shaker pulse rate, they may appear to be constantly illuminated. As the aircraft is recovered from the stall, the stall vane transducers will signal the computer which will reduce and finally remove pusher force. As long as the aircraft is operated at speeds requiring high indicated angles of attack, the control column shakers will continue to operate and the warning lights flash. The stall warning computer incorporates a stall rate function which senses and compensates for rapid angle-of-attack increases associated with abrupt pitch-up maneuvers. The rate circuits activate the pusher up to 4 knots prior to a stall in dynamic ratio to stall vane angular changes.

With power on, one or both Stall Warning Switches on, and squat switches in ground mode, setting the test switch to left or right test position activates test circuits which simulate a stall sequence as follows.

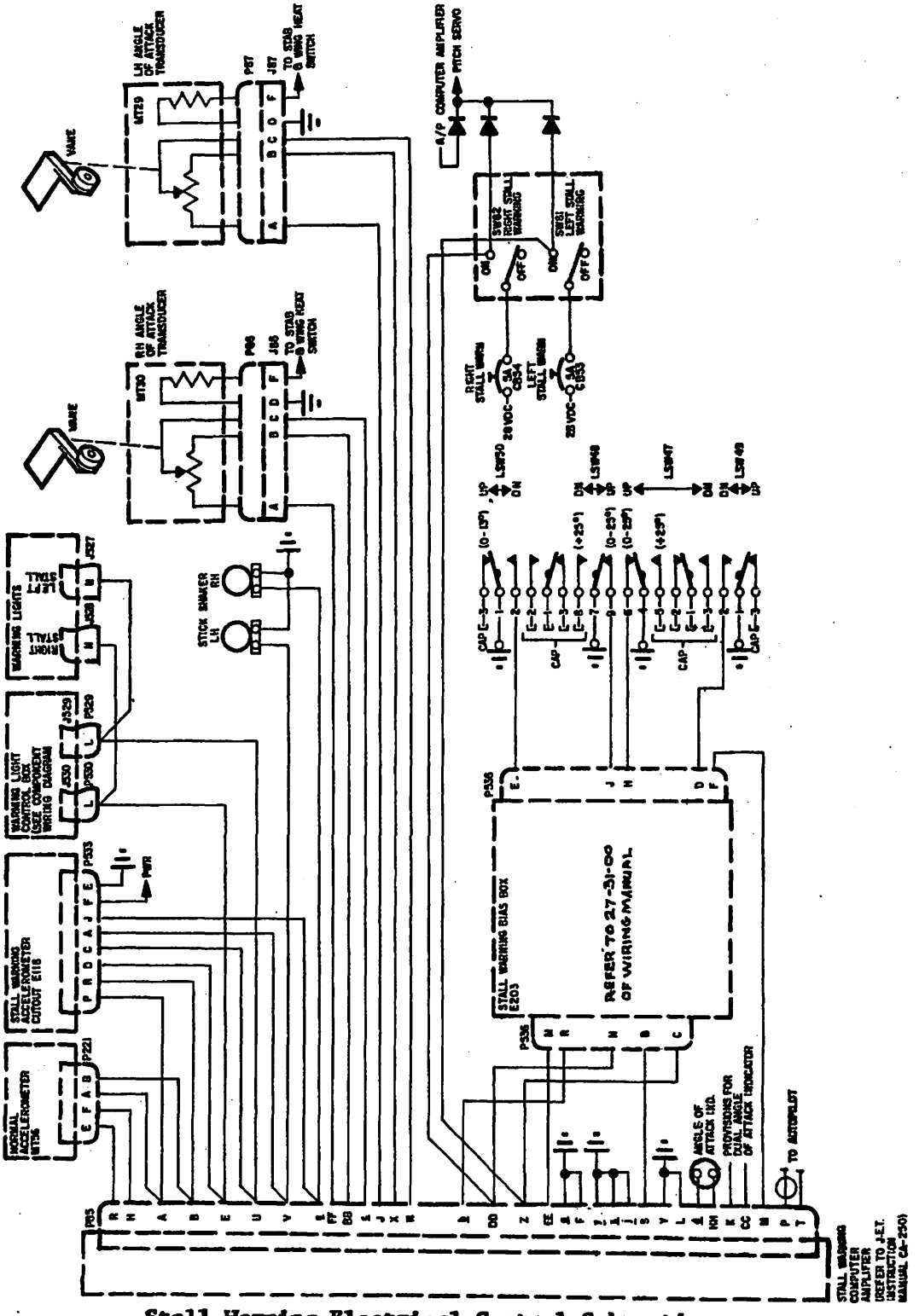
The indicator needle will start a sweep from the green band toward red line. As the needle enters the yellow band, the stall warning lights will flash and control column shakers will actuate. The shaker-warning light pulse rate will increase as the indicator needle travels toward red line. At red line, the pusher will pitch the column forward with shaker and warning light pulse rate at maximum level. Then pusher force will diminish and the indicator needle will fall back toward the yellow band. The test sequence of events is the same for both systems.

E. Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified for AMK 81-7 or AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement." (See figure 5.)

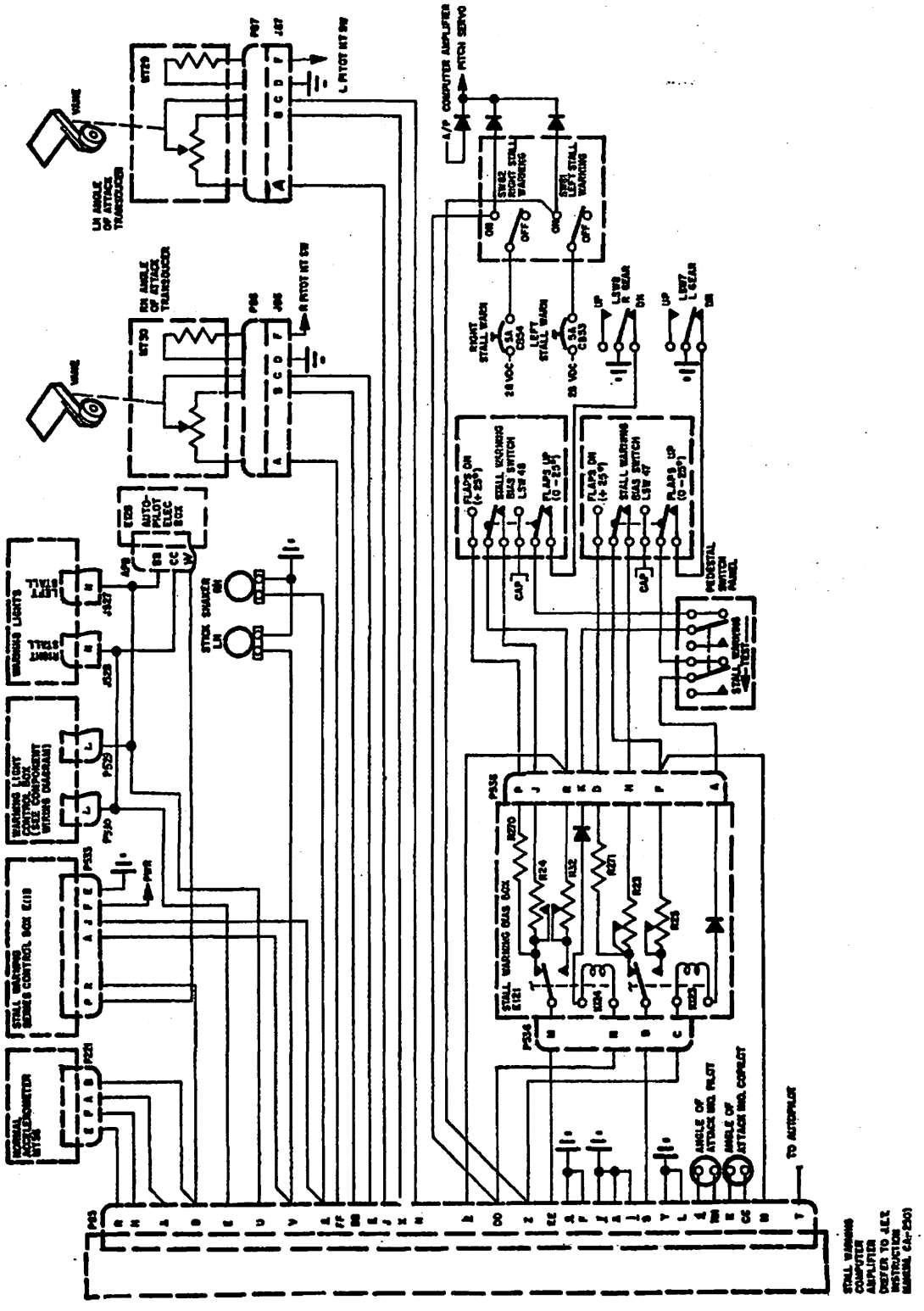
(1) Operation of the stall warning system is the same as described in paragraph 2.A., 2.B., 2.C., and 2.D. except that the control column nudger will actuate simultaneously with the shaker. The nudger is evidenced by a pulsing of the control column (1-1/2 to 2-1/2 times per second) forward (nose down) by the pitch servo. Pusher can be interrupted by depressing the control wheel master (MSW) Switch on either control wheel.



Stall Warning Electrical Control Schematic
Figure 1



Stall Warning Electrical Control Schematic
Figure 2



Stall Warning Electrical Control Schematic
Figure 3

STALL WARNING SYSTEM - TROUBLE SHOOTING

1. TROUBLE SHOOTING

A. Tools and Equipment

NAME	PART NUMBER	MANUFACTURER	USE
VOM (Analog type)	No. 260 or equivalent.	Simpson	Resistance and voltage checks.

B. Stall Warning System Trouble Shooting Ground Functional Test (Aircraft 25-206 and Subsequent and prior aircraft equipped with AAK 76-4, "Reduced Approach Speed System.")

CAUTION: THE FOLLOWING RESISTANCE CHECKS ARE TO BE PERFORMED WITH POWER REMOVED FROM THE AIRCRAFT. STEPS (7)(g), (h), AND (i) REQUIRE THE APPLICATION OF HYDRAULIC AND ELECTRICAL POWER TO THE AIRCRAFT. USE EXTREME CAUTION IN PERFORMING THESE STEPS.

NOTE: ° These procedures are provided as a trouble shooting guide only.

° The following check must be performed using an analog VOM. A noticeable change in resistance may be experienced if a digital VOM is used.

(1) Set Stall Warning Switches to OFF.
(2) Pull L STALL and R STALL circuits breakers.
(3) Ensure flaps are in full retracted position.
(4) Place aircraft on jacks. (Refer to Chapter 8.)
(5) Disconnect aircraft batteries.
(6) Remove nose compartment access doors. Disconnect electrical connectors from stall warning computer.
(7) Using an ohmmeter perform the following resistance checks:
 (a) Check pins G and H (P913) to aircraft ground. Ohmmeter reading shall be greater than 10K ohms. Record reading on Resistance Check Table, figure 101.
 (b) Check pins G, P, and H (P913) to aircraft ground. Ohmmeter reading shall show continuity. Record reading on Resistance Check Table, figure 101.
 (c) Check pins A, B, and C (P913) to aircraft ground. Ohmmeter shall show open circuit. Record reading on Resistance Check Table, figure 101.
 (d) Check pin F (P913) to aircraft ground. With squat switch in air mode (aircraft on jacks) ohmmeter reading shall be greater than 50 ohms. Activate LH squat switch to ground mode. Ohmmeter reading shall be less than 0.5 ohm. Record on Resistance Check Table, figure 101.

(e) Check pins A to C (P913). Ohmmeter reading shall be approximately 1.8K ohms. Move LH angle-of-attack vane. Ohmmeter reading shall not change. Record reading on Resistance Check Table, figure 101.
(f) Check pins B to C (P913). Ohmmeter reading shall be zero to approximately 1.8K ohms and shall vary with vane movement. Resistance shall decrease as vane is raised. Record reading on Resistance Check Table, figure 101.
(g) Check pin U (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps slightly, ohmmeter shall read less than 0.5 ohm. Record reading on Resistance Check Table, figure 101.
(h) Check pin E (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps past 13°, ohmmeter shall read less than 0.5 ohm. Record reading on Resistance Check Table, figure 101.
(i) Check pin F (P913) to aircraft ground. Ohmmeter shall read over 50 ohms with flaps fully retracted. Lower flaps past 25°, ohmmeter shall read less than 0.5 ohm. Record reading on Resistance Check Table, figure 101.
(j) Check pin Z (P913) to pin Z (P914). Ohmmeter shall show continuity.
(k) Check pin B (P913) to pin B (P914). Ohmmeter shall show continuity.

(8) Repeat steps (7)(a) through (7)(k) for RH Stall Warning System using electrical connector P914.
(9) Connect aircraft batteries and set Battery Switches to BAT 1 and BAT 2.
(10) Ensure that Stall Warning Switches are set to OFF.
(11) Depress L STALL, R STALL, and WARN LT circuit breakers.
(12) Ensure flaps are fully retracted.
(13) Using a voltmeter, perform the following voltage checks:
(a) Check pins E and R (P913) to aircraft ground. Voltmeter reading shall be zero volts. Set Stall Warning Switch to L. Voltmeter reading shall be 28 volts. Set L Stall Warning Switch to OFF. Record readings on Voltage Check Table, figure 102.
(b) Check pin C (P913) to aircraft ground. Voltmeter reading shall be zero volts. Set Stall Warning Switch to L, set system TEST Switch to L STALL, and depress pushbutton. Voltmeter reading shall be 28 volts. Set L Stall Warning switch to OFF. Record reading on Voltage Check Table, figure 102.
(14) Repeat steps (13)(a) and (13)(b) for the RH Stall Warning System using electrical connector P914.

NOTE: Resolve any wiring discrepancies before proceeding.

Para	Probe Pin	Desired Ohmmeter Reading	Actual Ohmmeter Reading	
			LH Stall System	RH Stall System
(7)(a)	G and H to Aircraft Ground	Greater than 10K ohms		
(7)(b)	G, P, and H to Aircraft Ground	Continuity		
(7)(c)	A, B, and C to Aircraft Ground	Open Circuit		
(7)(d)	F to Aircraft Ground	Squat Switches (Air Mode) - Greater than 50 ohms		
		Squat Switches (Gnd Mode) - Less than 0.5 ohm		
(7)(e)	A to C	Approximately 1.8K ohms		
		No change with vane movement		
(7)(f)	B to C	Zero to approx. 1.8K ohms, varies with vane position		
		Resistance decreases as vane is raised.		
(7)(g)	U to Aircraft Ground	Flaps Fully Retracted - Over 50 ohms		
		Flaps Slightly Lowered - Less than 0.5 ohm		
(7)(h)	E to Aircraft Ground	Flaps Fully Retracted - Over 50 ohms		
		Flaps Lowered past 13° - Less than 0.5 ohm		
(7)(i)	F to Aircraft Ground	Flaps Fully Retracted - Over 50 ohms		
		Flaps Lowered past 25° - Less than 0.5 ohm		
(7)(j)	Z (P913) to Z (P914)	Continuity		
(7)(k)	B (P913) to B (P914)	Continuity		

Resistance Check Table
Figure 101

(15) Connect electrical connectors to stall warning computer. Ensure that Stall Warning Switches are set to OFF. Set Battery Switches to BAT 1 and BAT 2. Verify that both L STALL and R STALL annunciators are illuminated (steady) and that both Master Warning annunciators are flashing.

(16) Set Battery Switches off and remove aircraft from jacks.

Para	Probe Pin	Desired Voltmeter Reading	Actual Voltmeter Reading	
			LH Stall System	RH Stall System
(13)(a)	E and R to Aircraft Ground	Stall Warning Switch off - Zero volts		
		Stall Warning Switch on - 28 volts		
(13)(b)	C to Aircraft Ground	Zero volts		
		Test Switch Actuated - 28 volts		

Voltage Check Table
Figure 102

STALL WARNING SYSTEM - MAINTENANCE PRACTICES

1. ADJUSTMENT/TEST

NOTE: ° Due to the possible effects on stall characteristic, whenever maintenance as noted in items (1) through (4) below is performed, the aircraft must be flight tested to verify aerodynamic acceptability and stall speeds. This flight test must be conducted by a pilot approved by Gates Learjet Corporation for stall test flights. The stall warning system must be checked in accordance with the flight or ground calibration procedures in 27-31-00 of this maintenance manual when maintenance as noted in items (1) through (5) is performed.

(1) When maintenance requiring removal and installation, repair, or installation of a new wing leading edge is performed. This includes loosening and retightening any of the leading edge screws.

(2) When a tip tank strake is loosened or removed from the tip tank or a new strake is installed. Applies to 25-206 and Subsequent and prior aircraft equipped with AAK 76-4, "Reduced Approach Speed System."

(3) When a new tip tank is installed. Applies to 25-206 and Subsequent and prior aircraft equipped with AAK 76-4, "Reduced Approach Speed System."

(4) When maintenance requiring repair to, or installation of, new stall strips is performed. Applies to 25-290 and Subsequent and prior aircraft equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers."

(5) When a stall warning transducer, computer, or indicator is repaired, replaced, or calibrated.

° On Aircraft 25-206 and Subsequent and prior Aircraft equipped with AAK 76-4, "Reduced Approach Speed System," replacement of the stall transducer vane is permitted in the event of a failed heater. Physical damage to the vane requires total transducer replacement because of possible damage to the transducer shaft, bearing, and potentiometer. Vane replacement requires functional test to verify that shaker and pusher actuation occurs within the prescribed tolerance of the vane plate marks.

° If angle-of-attack indicator replacement does not require an IND or R_x potentiometer adjustment and a quick check shows indicator to be at intersection of yellow/green for shaker and at intersection of red/yellow for pusher then the aircraft does not require flight test/adjustment or ground calibration adjustment.

° Perform functional test of stall warning system in accordance with the current inspection interval specified in Chapter 5.

NOTE: ° A flashing stall warning annunciator indicates proper system operation. A steady stall warning annunciator indicates system malfunction unless stall switches are off or angle of attack needle is in red area.

° Angle-of-attack vanes must be moved slowly to give correct indication. The rate of vane movement is used in stall computation. Failure to move vanes slowly will cause angle-of-attack indicator needle movement to be jumpy.

° Components of stall warning system that do not meet functional test requirements shall be replaced or recalibrated.

° The following paragraph references, with aircraft serials and modification definitions, are provided to aid maintenance personnel in determining the proper functional test procedures to use.

(1) Paragraph 1.A., Functional Test of Stall Warning System, applies to the following:

(a) Aircraft 25-061, 25-070 thru 25-205 NOT equipped with AAK 76-4, "Reduced Approach Speed System."

(b) Aircraft 25-061, 25-070 thru 25-205 equipped with or NOT equipped with AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."

(c) Aircraft 25-061, 25-070 thru 25-205 equipped with or NOT equipped with AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement."

These aircraft must comply with paragraph 1.A. in accordance with the current inspection interval specified in Chapter 5. If the aircraft stall warning system complies with paragraph 1.A., then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 1.A., then paragraphs 1.B. and 1.C. must be complied with.

(2) Paragraph 1.D., Functional Test of Stall Warning System, applies to the following:

(a) Aircraft 25-061, 25-070 thru 25-205 when certified for operation in Australia.

These aircraft must comply with paragraph 1.D. in accordance with the current inspection interval specified in Chapter 5. If the aircraft stall warning system complies with paragraph 1.D., then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 1.D., then paragraphs 1.E. and 1.F. must be complied with.

(3) Paragraph 1.G., Functional Test of Stall Warning System, applies to the following:

(a) Aircraft 25-206 thru 25-341, except 25-337 NOT equipped with AMK 81-7, "Horizontal Stabilizer Trim and Autopilot Improvement."

NOTE:

(b) Aircraft 25-061, 25-070 thru 25-205 equipped with AAK 76-4, "Reduced Approach Speed System."

(c) Aircraft 25-061, 25-070 thru 25-205 equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers."

These aircraft must comply with paragraph 1.G. in accordance with the current inspection interval specified in Chapter 5. If the aircraft stall warning system complies with paragraph 1.G., then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 1.G., then paragraphs 1.H., 1.K., 1.L., or 1.F. as applicable must be complied with.

(4) Paragraph 1.I., Functional Test of Stall Warning System, applies to the following:

(a) Aircraft 25-337, 25-342 and Subsequent.

(b) Aircraft 25-206 thru 25-341, except 25-337 equipped with AMK 81-7, "Horizontal Stabilizer Trim and Autopilot Improvement."

(c) Aircraft 25-061, 25-070 thru 25-205 when modified per AAK 76-4, "Reduced Approach Speed System" and AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement."

These aircraft must comply with paragraph 1.I. in accordance with the current inspection interval specified in Chapter 5. If the aircraft stall warning system complies with paragraph 1.I., then the aircraft can be returned to service. If the aircraft stall warning system does not comply with paragraph 1.I., then paragraphs 1.J., 1.L., or 1.M. must be complied with.

A. Functional Test of Stall Warning System

NOTE: ° The following functional test applies to:

(1) Aircraft 25-061, 25-070 thru 25-205 NOT equipped with AAK 76-4, "Reduced Approach Speed System" or AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."

(2) Aircraft 25-061, 25-070 thru 25-205 equipped with AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."

(3) Aircraft 25-061, 25-070 thru 25-205 equipped with or NOT equipped with AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement."

° This test must be performed in accordance with the current inspection intervals specified in Chapter 5.

CAUTION: AVOID CONTACT WITH HOT VANES.

(1) Connect auxiliary electrical power and hydraulic power to aircraft. Set Battery Switch(es) on.

(2) Set Pitot Heat Switches to L and R and verify that vanes are heated. Set Pitot Heat Switches to OFF.

(3) Set applicable Stall Warning Switch on. Allow 15-minutes warm-up time for stall warning computer.
(4) Hold control wheel and observe stall warning lights while a second man aids in performing the following:

NOTE: On Aircraft 25-061, 25-070 thru 25-205 modified per AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement," control column nudger will actuate simultaneously with the shaker. The nudger is evidenced by a pulsing of the control column (1-1/2 - 2-1/2 times per second) forward (nose down) by the pitch servo.

(a) Extend flaps to full down (40°).
(b) Slowly lift angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●●●. Shaker shall actuate and left stall warning light shall flash.
(c) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ●●●. Pusher shall actuate and left stall warning light shall flash.
(d) Retract flaps to full up (0°). On Aircraft 25-199 thru 25-205, and prior aircraft modified per AAK 75-5, "Installation of Components for Partial Flap Extension During Takeoff - 8° Flap System," retract flaps to 20°.
(e) Lift left angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●●. Shaker shall actuate and left stall warning light shall flash.
(f) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ●●. Pusher shall actuate and left stall warning light shall flash.
(g) With flaps up, hold Stall Warning Test Switch on pedestal to TEST for steps (h) and (i).

NOTE: On aircraft equipped with 8° flap system, the test switch is not installed.

(h) Lift LH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●. Shaker shall actuate and left stall warning light shall flash.
(i) Continue to lift the LH angle-of-attack vane until aft upper edge of vane is centered on upper line marked ●. Pusher shall actuate and left stall warning shall flash. Release test switch.
(j) Repeat steps (4) through (4)(i) using RH stall vane and RH Stall Warning Switch.
(k) Set Battery and Stall Warning Switches to OFF.
(l) Check upper travel limit of angle-of-attack vanes as shown in figure 201.
(m) If the stall warning system complies with this functional test, perform step (o) and return aircraft to service.

(n) If stall warning system fails to comply with this functional test, then the stall warning system ground adjustment (paragraph 1.B.) and the stall warning system flight test/adjustment (paragraph 1.C.) must be performed.

(o) Set Battery Switch(es) and Stall Warning Switches off. Install cover on computer. Install nose compartment doors. Remove aircraft from jacks.

B. Stall Warning System Ground Adjustment (See figure 201.)

NOTE: ° The following ground adjustment is applicable to:

(1) Aircraft 25-061, 25-070 thru 25-205 NOT equipped with AAK 76-4, "Reduced Approach Speed System" or AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."

(2) Aircraft 25-061, 25-070 thru 25-205 equipped with AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."

(3) Aircraft 25-061, 25-070 thru 25-205 equipped with or NOT equipped with AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement."

° The ground adjustment of this paragraph is a preliminary calibration to the required flight test/adjustment.

° The pusher-shaker should be adjusted only by completely qualified personnel, preferably at a factory authorized service center. It will be necessary to fabricate patch cables in order for the computer and the stall warning bias box to be accessible in the cabin during the flight required by paragraph 1.C.

° Use a straightedge when aligning the angle-of-attack vanes with marks on the vane plates.

° Verify that flap position switches and landing gear squat switches are correctly adjusted.

(1) Remove nose compartment access covers.
(2) Connect auxiliary electrical and hydraulic power sources to aircraft.
(3) Potentiometers R23 through R25 and R32 are externally adjustable on the stall warning bias box. Adjust R23, R24, R25, and R32 fully clockwise.
(4) Remove cover from computer (end opposite plug). Using a small screwdriver, turn left and right PUSH, SHAKE, and IND poteniometers clockwise until they click.
(5) Set Battery Switch(es) and LH Stall Warning Switch to ON. Allow 15-minutes warm-up time for stall warning computer.
(6) With a man in the cockpit, hold control wheel and observing stall warning lights and angle-of-attack indicator, perform the following:

NOTE: On Aircraft 25-061, 25-070 thru 25-205 modified per AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement," control column nudger will acuate simultaneously with the shaker. The nudger is evidenced by a pulsing of the control column (1-1/2 to 2-1/2 times per second) forward (nose down) by the pitch servo.

(a) Extend flaps to full down (40°).
(b) Lift LH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●●●. Turn left SHAKE potentiometer counterclockwise until shaker actuates and left stall waring light flashes.
(c) Continue to left angle-of-attack until aft upper edge of vane is centered on upper line marked ●●●. Turn left PUSH potentiometer counterclockwise until pusher actuates and left stall warning light flashes. Turn right IND potentiometer counterclockwise until angle-of-attack indicator needle is at intersection of amber and red bands. Move angle-of-attack vane in shaker position; angle-of-attack indicator needle will be within proximity of intersection between green and amber bands.

NOTE: Repeat steps (b) and (c) to stabilize each adjustment.

(d) Retract flaps to full up (0°). On Aircraft 25-199 thru 25-205, and prior Aircraft modified per AAK 75-5, "Installation of Components for Partial Flap Extension during Takeoff - 8° Flap System," retract flaps to 20°. Check with a protractor. Do not rely on position indicator.
(e) Lift LH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●●. Turn potentiometer R25 or RS25 counterclockwise until shaker and left stall warning light flashes.
(f) Continue to lift angle-of-attack vane until aft upper edge is centered on upper line marked ●●. Pusher shall actuate and left stall warning light shall flash.
(g) With flaps up, hold Stall Warning Test Switch (on pedetal switch panel) to TEST for steps (h) and (i).

NOTE: On Aircraft with 8° flap system, the test switch is not installed.

(h) Lift LH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●. Turn potentiometer R23 or RS23 counterclockwise, until shaker actuates and left stall warning light flashes.
(i) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ●. Pusher shall actuate and left stall warning light shall flash.

(j) Set LH Stall Warning Switch to OFF and RH Stall Warning Switch to ON.
(k) Extend flaps to full down (40°).
(l) Lift RH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●●● . Turn right SHAKE potentiometer counterclockwise until shaker actuates and right stall warning light flashes.
(m) Continue to lift RH angle-of-attack vane until aft upper edge of vane is centered on upper line marked ●●● . Turn right PUSH potentiometer counterclockwise until pusher actuates and right stall warning light flashes. Turn left IND potentiometer counterclockwise until angle-of-attack indicator needle is at intersection of amber and red bands. Move angle-of-attack vane to shaker position; angle-of-attack indicator needle will be within proximity of intersection of green and amber bands.

NOTE: Repeat steps (l) and (m) to stabilize each adjustment.

(n) Retract flaps to full up (0°). On Aircraft 25-199 thru 25-205, and prior Aircraft modified per AAK 75-5, "Installation of Components for Partial Flap Extension during Takeoff - 8° Flap System," retract flaps to 20°. Check with a protractor. Do not rely on position indicator.
(o) Lift RH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●● . Turn potentiometer R32 or RS26 counterclockwise until shaker actuates and right stall warning light flashes.
(p) Continue to lift angle-of-attack vane until aft upper edge is centered on upper line marked ●● . Pusher shall actuate and right stall warning light shall flash.
(q) Retract flap to full up and hold Stall Warning Test Switch (on pedestal switch panel) to TEST for steps (r) and (s).

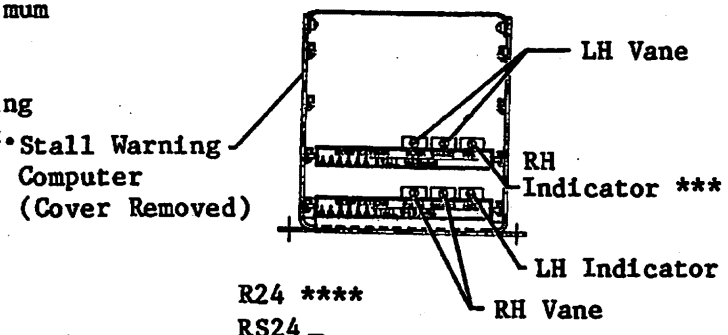
NOTE: On aircraft equipped with 8° flap system, the test switch is not installed.

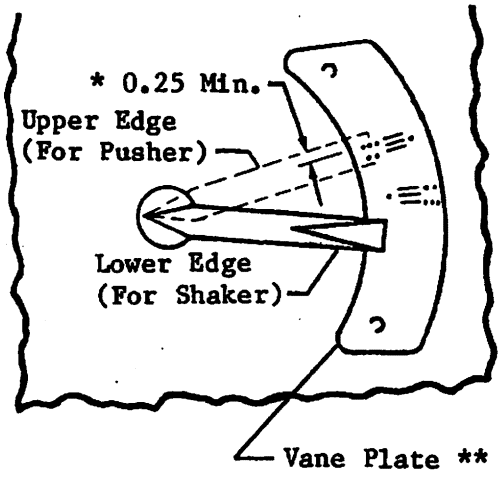
(r) Lift RH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● . Turn potentiometer R24 or RS24 counterclockwise until shaker actuates and right stall warning light flashes.
(s) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ● . Pusher shall actuate and right stall warning light shall flash.
(t) Check voth vanes for upper travel limit as shown. (See figure 201.) If adjustment is necessary, remove transducer cover and rotate adjusting screw until upper edge (for pusher) of vane stops a minimum of 0.25 inch above highest mark on plate. Reinstall transducer cover and safety. Apply ProSeal 727 around pigtail at left transducer cover and safety.
(u) Set Battery and RH Stall Warning Switches to OFF.

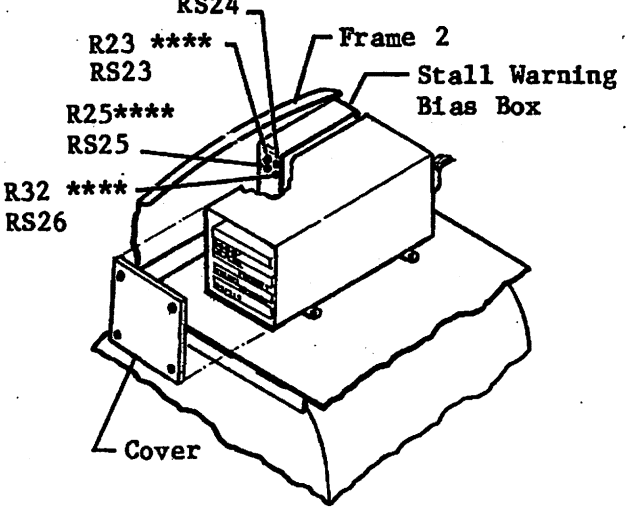
* Upper travel limit of vane: Minimum of 0.25 inch above pusher.

** The sequence of vane plate marking may vary with different aircraft.

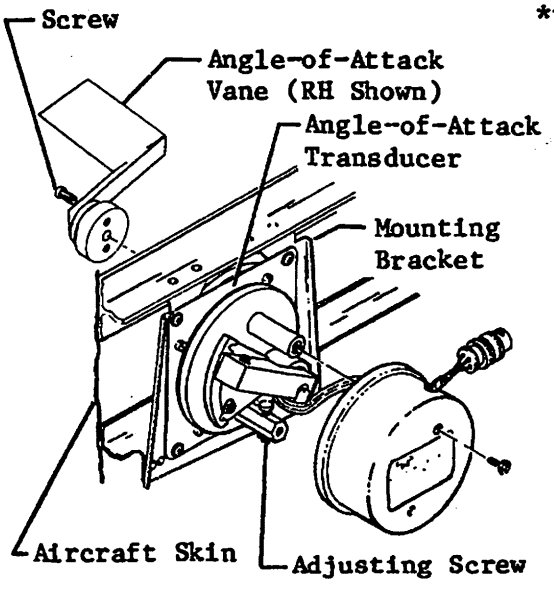
*** The RH indicator is optional equipment and may not be installed on some aircraft.





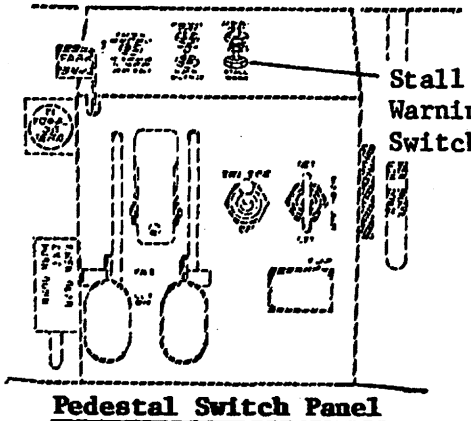


**** Aircraft 25-199 thru 25-205 and prior Aircraft modified per AAK 75-5 (Installation of Components for Partial Flap Extension During Takeoff-8° Flap System).



ANGLE OF ATTACK VANE INSTALLATION

PUSHER, SHAKER, AND INDICATOR ADJUSTMENT POTENTIOMETERS



Pedestal Switch Panel

Stall Warning System Adjustment
Figure 201

C. Stall Warning System Flight Test/Adjustment (See figure 201.)

NOTE: The following flight test/adjustment is applicable to:
(1) Aircraft 25-061, 25-070 thru 25-205 NOT equipped with AAK 76-4, "Reduced Approach Speed System" or AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."
(2) Aircraft 25-061, 25-070 thru 25-205 equipped with AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."
(3) Aircraft 25-061, 25-070 thru 25-205 equipped with or NOT equipped with AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement."

CAUTION: STALL WARNING SYSTEM FLIGHT TESTING MUST BE PERFORMED BY A PILOT APPROVED BY GATES LEARJET CORPORATION FOR STALL WARNING CALIBRATION.

(1) Preflight
(a) Fabricate patch cables in accordance with figure 202. The cables must be wired exactly as plug P85 on the computer-amplifier and plug P635 on the Stall Warning Bias Box. Refer to applicable wiring diagram in the Wiring Manual.

NOTE: There will be more wires in the patch cable bundle than there are pins in the frame No. 5 connector (3). Therefore some of the ground may be grouped by "jumpering" to reduce the quantity of wires to a number which the connector will accommodate.

(b) Remove the plug and plate from the patch cable access hole in aircraft frame No. 5. This hole is located approximately 5 inches directly below the outflow valve on most aircraft.
(c) Remove the Stall Warning Bias Box and the computer from the nose compartment and place in cabin area accessible for flight crew adjustments.
(d) Connect wire bundles in nose compartment to computer and bias box in cabin area with patch cables.
(e) Perform stall warning and nose steering system functional check to prove integrity of patch cables.
(f) Fuel the aircraft wing and tip tanks.
(g) Weigh the aircraft. Add ballast as required so that the C.G. is at the forward limit as shown on the Stall Speed Charts and Airplane Flight Manual C.G. chart.

(2) Flight Check at 8,000 to 12,000 Feet MSL (5000 Foot AGL minimum) at forward CG (forward to 3% aft of forward).

NOTE: The following flight check is applicable to:
(1) Aircraft 25-061, 25-070 thru 25-205 NOT equipped with AAK 76-4, "Reduced Approach Speed" or AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."
(2) Aircraft 25-061, 25-070 thru 25-205 equipped with or NOT equipped with AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement."

(a) In landing, takeoff, and cruise configuration, respectively, with idle power, approach stall by decreasing airspeed at one (1) knot per second. Check each system separately.
(b) Record fuel quantity and airspeeds at which shaker and pusher actuates.
(c) Verify proper operation of stall warning "steady" and "flashing" lights.
(d) Pusher actuation speeds shall fall on the stall speed curve +2/-0 knots for each configuration. Refer to FAA Approved Flight Manual (Section IV) for Stall Speeds. Enter Stall Speed chart using instrument corrected airspeed as provided by the correction card for the Pilot's Airspeed Indicator.
(e) The shaker shall actuate at least 7% (in no case less than 7 knots) prior to the pusher actuation in each configuration.
(f) Angle-of-attack needle shall enter red arc at pusher activation.
(g) Adjust system as required until requirements of steps (d) and (f) above are met.
(h) Recheck system. Pusher speed shall be within +2/-0 knots of speeds reflected in the Approved Airplane Flight Manual stall speed charts.

(3) Flight Check at 8,000 to 15,000 Feet MSL (5000 Foot AGL minimum) at forward CG (forward to 1.54% aft of forward). Aircraft 25-061, 25-070 thru 25-205 equipped with AAK 82-8 or AMK 83-4, "Handling Qualities Improvement."
(a) In landing, takeoff, and cruise configuration, respectively, with idle power, approach stall by decreasing airspeed at one (1) knot per second. Check each system separately.
(b) Check aerodynamic stall at 12,000 feet. Roll shall not exceed 20° and speed shall not exceed pusher chart speed.
(c) Record fuel quantity and airspeeds at which shaker and pusher actuates.
(d) Verify proper operation of stall warning "steady" and "flashing" lights.
(e) Pusher actuation speeds shall fall on the stall speed curve ±3 knots for each configuration. Refer to FAA Approved Flight Manual (Section IV) for Stall Speeds. Enter Stall Speed chart using instrument corrected airspeed as provided by the correction card for the Pilot's Airspeed Indicator.

(f) The shaker shall actuate at least 7% (in no case less than 7 knots) prior to the pusher actuation in each configuration.
(g) Angle-of-attack needle shall enter red arc at pusher activation.
(h) Adjust system as required until requirements of steps (d) and (f) above are met.
(i) Recheck system. Pusher speed shall be within ±3 knots of speeds reflected in the Approved Airplane Flight Manual stall speed charts.

(4) Postflight
(a) Remove patch cables from aircraft. Install computer and bias box in nose compartment.
(b) Install plug and plate in frame 5 access hole. Seal over pressure side of plate with ProSeal 890B.

CAUTION: DO NOT PERFORM ANY ADJUSTMENT OF COMPUTER PUSHER AND SHAKER ACTUATION POINTS AFTER COMPLETION OF THE FLIGHT ADJUSTMENT.

(c) Repeat "Functional Test of Stall Warning System," paragraph 1.A., to verify that the existing vane plate markings for pusher and shaker match the flight-calibrated computer. If the vane plate markings match the computer, return aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to precisely match the computer pusher and shaker actuation points. (Refer to 27-31-10.)

D. Functional Test of Stall Warning System (Aircraft Certified for Operation in Australia.)

NOTE: This test must be performed in accordance with current inspection intervals specified in Chapter 5.

CAUTION: AVOID CONTACT WITH HOT VANES.

(1) Set Battery Switch(es) on.

Use components noted below to make up noted plugs. All components are manufactured by AMP, Inc., except for connector (3) which is manufactured by Deutsch Electronics Corp.

(1) Plug
201358-1 Housing
201389-2 Jackscrew (F)
42881-1 Sockets
201182-1 Clamp

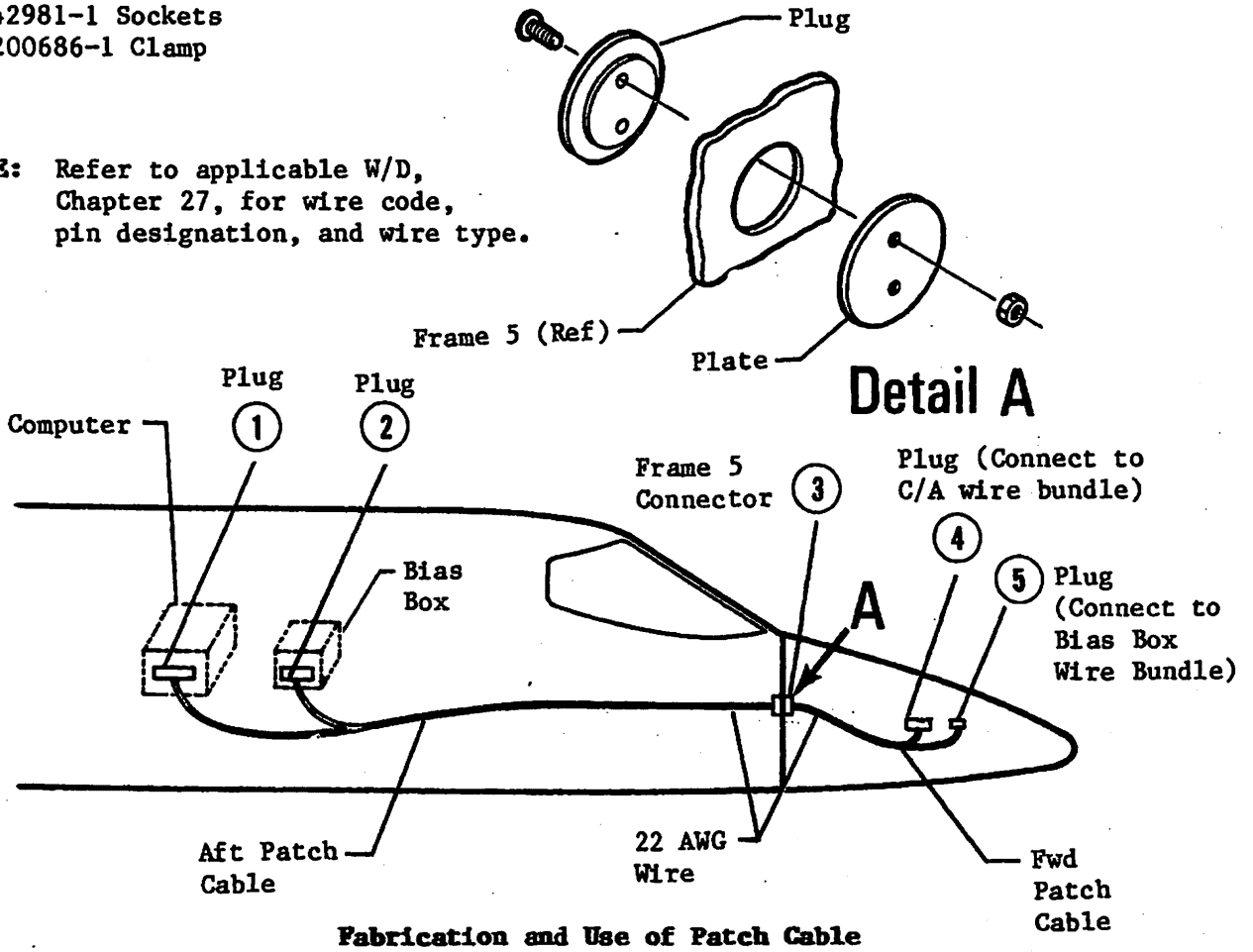
(2) Plug
201355-1 Housing
201388-2 Jackscrew (M)
201389-2 Jackscrew (F)
42981-1 Sockets
200686-1 Clamp

(3) Connector
RSM04-19-33S Receptacle
RSM07-19-33P Plug

(4) Plug
200277-2 Housing
200874-2 Jackscrew (M)
42980-1 Pins
201390-2 Pin Guard

(5) Plug
201298-1 Housing
200874-2 Jackscrew (M)
200875-2 Jackscrew (F)
42980-1 Pins
201363-4 Pin Guard

NOTE: Refer to applicable W/D, Chapter 27, for wire code, pin designation, and wire type.



Fabrication and Use of Patch Cable
Figure 202

(2) Set Pitot Heat Switches to L and R and verify that vanes are heated. Set Pitot Heat Switches to OFF.
(3) Set both Stall Warning Switches on. Allow 15-minute warm-up time for stall warning computer.
(4) Hold control wheel and observe stall warning lights while a second man aids in performing the following:

NOTE: ° The pitch torquer will not actuate until one vane has reached the pusher actuation point and the other vane has actuated its stick shaker.

° Stall Warning lights illuminate at pusher actuation. Aural warning sounds at the shaker actuation.

° Each system will flash both stall warning lights when its vane reaches the pusher actuation point, if both system are turned on. If one system is turned off, its light will burn steady and not be flashed by the other system.

(a) Extend flaps to full down (40°).
(b) Lift LH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● ● ● . Shaker shall actuate and aural warning shall sound.
(c) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ● ● ● . Stick shaker and aural warning shall continue and both stall warning lights shall flash.
(d) Lift RH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● ● ● (RH shaker actuation point). Pusher shall actuate, both stall warning lights shall continue to flash and stick shaker and aural warning shall continue to function.
(e) Repeat steps (b) thru (d) using the opposite angle-of-attack vanes.
(f) Retract flaps (above 20°).
(g) Lift LH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● ● . Shaker shall actuate and aural warning shall sound.
(h) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ● ● . Stick shaker and aural warning shall continue and both stall warning lights shall flash.
(i) Lift RH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● ● (RH shaker actuation point). Pusher shall actuate, both stall warning lights shall continue to flash and stick shaker and aural warning shall continue to function.
(j) Repeat steps (g) thru (i) using the opposite angle-of-attack vanes.
(k) With flaps up, hold Stall Warning Test Switch ON for steps (1) thru (n).

NOTE: The Stall Warning Test Switch simulates a gear up condition.

(l) Lift LH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● . Shaker shall actuate and aural warning shall sound.
(m) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ● . Stick shaker and aural warning shall continue and both stall warning lights shall flash.
(n) Lift RH angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● (RH shaker actuation point). Pusher shall actuate, both stall warning lights shall continue to flash and stick shaker and aural warning shall continue to function.
(o) Repeat steps (1) thru (n) using the opposite angle-of-attack vanes.
(p) Set Battery and Stall Warning Switches to OFF.
(q) Check upper travel limit of angle-of-attack vanes as shown.
(r) If the stall warning system complies with this functional test, perform step (t) and return aircraft to service.
(s) If stall warning system fails to comply with this functional test, then the stall warning system ground adjustment (paragraph 1.E.) and the stall warning system flight test/adjustment (paragraph 1.F.) must be performed.
(t) Set Battery Switch(es) and Stall Warning Switches off. Install cover on computer. Install nose compartment doors. Remove aircraft from jacks.

E. Stall Warning System Ground Adjustment (Aircraft Certified for operation in Australia.) (See figure 201.)

NOTE: ° The ground adjustment of this paragraph is a preliminary calibration to the required flight test/adjustment.

° The pusher-shaker systems should be adjusted only by completely qualified personnel, preferably at a factory authorized service center. It will be necessary to fabricate patch cables in order for the computer and the stall warning bias box to be accessible in the cabin during the flight required by paragraph 1.F.

° Use a straightedge when aligning the angle-of-attack vanes with marks on the vane plates.

° Verify that flap position switches and landing gear squat switches are correctly adjusted.

(1) Remove nose compartment access doors.
(2) Attach an auxiliary hydraulic unit to aircraft.

(3) Potentiometers RS23 through RS26 are externally adjustable on stall warning bias box. Turn RS23 through RS26 potentiometers fully clockwise.

(4) Remove cover from nose steering computer (end opposite plug). Using a small screwdriver, turn left and right PUSH, SHAKE, and IND potentiometers fully clockwise until they click.

(5) Set Battery Switch(es) and both Stall Warning Switches to ON. Allow 15-minute warm-up time for stall warning computer.

(6) Hold control wheel and observe stall warning lights and angle-of-attack indicator while second man aids in performing the following:

(a) Extend flaps to full down (40°).

(b) Lift left angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●●●. Turn left SHAKE potentiometer counterclockwise until shaker actuates and aural warning sounds. Position vane fully down.

(c) Repeat step (b) for RH vane using RH SHAKE potentiometer.

(d) Hold RH vane at shaker actuation, and continue to lift left angle-of-attack vane until aft upper edge of vane is centered on upper line marked ●●● . Turn left PUSH potentiometer counterclockwise until both stall warning lights flash and pusher actuates. Stick shaker and aural warning shall continue to function. Turn left IND potentiometer counterclockwise until pilot's angle-of-attack indicator needle is at intersection of amber and red bands. Move angle-of-attack vane to shaker position; pilot's angle-of-attack indicator needle shall be within proximity of intersection of green and amber bands.

(e) Repeat steps (b) and (d) to stabilizer each adjustment.

(f) Repeat steps (b) and (d) using right angle-of-attack vane and right IND potentiometer for copilot's angle-of-attack indicator. Repeat steps (b) and (d) to stabilize each adjustment.

(g) Retract flaps (above 20°). Check with a protractor. Do not rely on position indicator.

(h) Lift left angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●● . Turn potentiometer RS25 counterclockwise until stick shaker actuates and aural warning sounds.

(i) Continue to lift angle-of-attack vane until aft upper edge is centered on upper line marked ●● . Both stall warning lights shall flash. Stick shaker and aural warning shall continue to function.

(j) Lift right angle-of-attack vane until aft lower edge of vane is centered on lower line marked ●● . Pusher shall actuate, both stick shakers and aural warning shall continue to function, and stall warning light shall continue to flash.

(k) Repeat steps (h) and (j) using right angle-of-attack vane and RS26 potentiometer.

(l) With flaps up, hold Stall Warning Test Switch to ON for steps (m) and (n).

NOTE: The Stall Warning Test Switch will simulate a gear up condition.

(m) Lift left angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● . Turn potentiometer RS23 counterclockwise until shaker actuates and aural warning sounds.
(n) Continue to lift angle-of-attack vane until aft upper edge of vane is centered on upper line marked ● . Both stall warning lights shall flash and stick shaker and aural warning shall continue to function.
(o) Lift right angle-of-attack vane until aft lower edge of vane is centered on lower line marked ● . Pusher shall actuate, both stick shakers and aural warning shall continue to function, and stall warning lights will continue to flash.
(p) Repeat steps (m) and (o) using right angle-of-attack vane and RS24 potentiometer.
(q) Set Battery and Stall Warning Switches off.
(r) Check both vanes for upper travel limit as shown. If adjustment is necessary, remove transducer cover and rotate adjusting screw until upper edge (for pusher) of vane stops 0.25 (±0.06) inch above highest mark on plate. Reinstall transducer cover and safety. Apply ProSeal 727 around pigtail at transducer opening, left transducer only.

F. Stall Warning System Flight Adjustment (Aircraft Certified for Operation in Australia.) (See figure 202.)

(1) Fabricate patch cables in accordance with figure 202. Cables must be wired exactly as plug P85 on the computer and plug P536 on stall warning bias box.

NOTE: There will be more wires in the patch cable bundle than there are pins in the frame No. 5 connector (3). Therefore, some ground may be grouped by "jumpering" to reduce the number of wires to a quantity which the connector will accommodate.

(2) Remove plug and plate from patch cable access hole in aircraft frame No. 5. This hole is located approximately 5 inches directly below the outflow valve on most aircraft.
(3) Remove stall warning bias box and computer from nose compartment and place in cabin area accessible for flight crew adjustments.
(4) Connect wire bundles in nose compartment to computer and bias box in cabin area with patch cables.
(5) Secure nose compartment access doors.
(6) Remove cover from computer (end opposite plug).
(7) Perform Stall Warning System Functional Test to prove integrity of patch cables.

(8) Record aircraft takeoff weight. Approach stall by decreasing airspeed at one (1) knot per second. Check each system separately. Fly the aircraft in each of the following configurations.

Gear Down - Flaps Down
Gear Down - Flaps Up
Gear Up - Flaps Up

(9) Record fuel quantity and airspeeds at which shaker pusher actuates. Using recorded fuel quantity and takeoff weight, calculate aircraft weight at shaker pusher actuation.
(10) Verify proper operation of stall warning "steady" and "flashing" lights.
(11) Pusher actuation speed shall fall on stall speed curve for each configuration. Refer to Approved Airplane Flight Manual (Section IV) for stall speeds. Enter stall speed charts using instrument-corrected airspeed as provided by correction card for Pilot's Airspeed Indicator.
(12) Shaker shall actuate at least 7% (in no case less than 7 knots) prior to pusher actuation with gear and flaps fully down.
(13) Angle-of-attack needle should enter red arc at pusher actuation.
(14) Adjust system as required until requirements of steps (11), (12), and (13) above are met.
(15) Recheck system. Pusher speed shall be within +2, -0 knots of speeds reflected in the Approved Airplane Flight Manual stall speed charts.

NOTE: When checking left or right stall systems separately, the stall warning lights will flash at the pusher point, but the pusher will not activate.

(16) Remove patch cables from aircraft and install computer and bias box in nose compartment.
(17) Install plug and plate in frame 5 access hole. Seal over pressure side of plate with ProSeal 890B.

CAUTION: DO NOT PERFORM ANY ADJUSTMENT OF COMPUTER PUSHER AND SHAKER ACTUATION POINTS AFTER COMPLETION OF THE FLIGHT ADJUSTMENT.

(18) Repeat "Functional Test of Stall Warning System," paragraph 1.D., to verify that the existing vane plate markings for pusher and shaker match the flight-calibrated computer. If the vane plate markings match the computer, return aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to precisely match the computer pusher and shaker actuation points. (Refer to 27-31-10.)

G. **Functional Test of Stall Warning System** (Aircraft 25-206 thru 25-341, except 25-337 and earlier aircraft equipped with AAK 76-4, "Reduced Approach Speed System" or AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips and Boundary Layer Energizers" but NOT modified per AMK 81-7, "Horizontal Stabilizer Trim and Autopilot Improvement.)

NOTE: This functional test must be performed to comply with the current inspection interval specified in Chapter 5.

(1) Connect electrical power to aircraft.

CAUTION: AVOID CONTACT WITH HOT VANES.

(2) Set Pitot Heat Switches on and verify that vanes are heated. Set Pitot Heat Switches off.

(3) With stall vanes completely lowered, set Stall Warning Switches on. Allow 15-minute warm-up time for stall warning computer. Verify that the following occurs:
 (a) L STALL and R STALL annunciators shall extinguish.
 (b) Pilot's and copilot's angle-of-attack indicator needle shall move into the green area.
 (c) Neither shaker nor pusher shall actuate, even transiently (a slight tremor at the control wheel is acceptable).

(4) Set Stall Warning Switches off. L STALL and R STALL annunciators shall illuminate.

(5) Actuate both squat switches to the air mode (aircraft on jacks) and set Stall Warning Switches on. Verify flaps at 0°.

(6) Set Stall Warning Test Switch (on pedestal switch panel) to L STALL. No response shall be noted.

(7) Set Stall Warning Test Switch (on pedestal switch panel) to R STALL. No response shall be noted.

(8) Actuate squat switches to ground mode. Set Stall Warning Test Switch to L STALL. Verify that the following occurs:
 (a) Pilot's angle-of-attack indicator needle shall start moving from green area towards the red area at a slow rate.
 (b) As the indicator needle moves into the yellow area, the L STALL annunciator shall flash and the LH stick shaker shall actuate (there may be a light lag as the shaker motor comes up to speed).
 (c) As the indicator needle reaches the red line, the L STALL annunciator flashing shall become steady momentarily, pusher shall actuate briefly and automatically stop as the indicator needle falls back into the yellow area. Set Stall Warning Test Switch off and verify that indicator needle falls back into the green area.

(9) Repeat steps (8)(a) through (8)(c) for the RH stall warning system with the Stall Warning Test Switch set to R STALL.

(10) Set Stall Warning Switches off. Place both stall vanes in the full down position.

(11) Remove attaching parts securing accelerometer to its mounting. Hold accelerometer in its normal position. Use caution when handling the accelerometer to prevent damage to the wiring.
(12) Set Stall Warning Switches on. Raise LH stall vane until pusher activates. Hold LH stall vane at this point. With pusher operating, slowly rotate accelerometer about its cylindrical axis until pusher force drops to approximately 1/2. Hold accelerometer in this position.
(13) Slowly raise RH stall vane until copilot stick shaker actuates. Verify that pusher force increases and continues to increase as RH vane is raised.
(14) Lower both stall vanes and return accelerometer to its normal position.
(15) Repeat steps (12) and (13) using the RH stall vane at pusher and LH stall vane at shaker.
(16) Set Stall Warning Switches off and install accelerometer in aircraft.
(17) Connect pitot static tester to the pitot and static systems. (Refer to Chapter 34.)
(18) Set Stall Warning Switches on. Ensure that both stall vanes are lowered to position the angle-of-attack indicator needle in the green band. Ensure that flaps are fully retracted.
(19) Slowly apply vacuum to the system and observe the pilot's and copilot's angle-of-attack indicators. At 22,500 (±750) feet, both indicators shall abruptly move toward the yellow area.
(20) Slowly release vacuum from system and remove pitot-static tester.
(21) Extend flaps full down.
(22) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ●●●●. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(23) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line marked ●●●●. The shaker shall actuate and the L STALL warning light will flash and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(24) Retract flaps to 20°.
(25) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ●●●. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(26) Slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked ●●●. The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(27) Retract flaps to 8°.
(28) Slowly lift left angle-of-attack vane until the upper edge of vane is centered on the upper line marked ●●. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.

(29) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line marked ●● . The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.

(30) Retract flaps to full up (assure flaps are all the way up).

(31) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ● . The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.

(32) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on lower line marked ● . The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.

(33) Repeat steps (21) through (32) using the right stall warning switch, right angle-of-attack vane, copilot's angle-of-attack indicator, and R STALL warning light.

(34) If the stall warning system complies with this functional test, perform step (36) and return the aircraft to service.

(35) If the stall warning system fails to comply with this functional test, proceed as follows:

(1) On Aircraft 25-206 thru 25-289 not equipped with AAK 79-10 or AMK 83-5, "Stall Fence," the aircraft must perform the "Stall Warning System Ground Adjustment" per paragraph 1.H. and the "Stall Warning System Flight Adjustment/Test" per paragraph 1.K.

(2) On Aircraft 25-290 thru 25-341, except 25-337 and prior aircraft equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers" but NOT modified per AMK 81-7, "Horizontal Stabilizer Trim and Autopilot Improvement," may elect to either (1) perform the "Stall Warning System Ground Adjustment" per paragraph 1.H. and the "Stall Warning System Flight Adjustment/Test" per paragraph 1.L. or, (2) perform the "Stall Warning System Ground Calibration" per paragraph 1.M.

(36) Set Battery Switches and Stall Warning Switches to off. Install end cover on computer. Install nose compartment doors. Remove aircraft from jacks.

H. Stall Warning System Ground Adjustment (Aircraft 25-206 thru 25-341, except 25-337 and earlier aircraft equipped with AAK 76-4, "Reduced Approach Speed System" or AAK 79-10 or AMK 83-5, "Installation of Wing Fence, Stall Strips, and Boundary Layer Energizers.") (See figure 203.)

NOTE: ° The pusher-shaker systems should be adjusted only by completely qualified personnel, prefereably at a factory authorized service center.

° Any adjustment of the pusher will require readjustment of the corresponding shaker point.

° Stall vanes must be moved very slowly to give correct indication because vane rate is used in stall computation.

(1) Remove nose compartment access doors.
(2) Remove end cover from computer to gain access to adjustment potentiometers.
(3) Connect auxiliary power and hydraulic units to aircraft.
(4) Retract flaps to full up.
(5) Set Stall Warning Switches off.
(6) Set Battery Switches on.
(7) Assure that both angle-of-attack vanes are full down.
(8) Set left Stall Warning Switch on. Allow 15-minute warm-up time for stall warning computer.
(9) Slowly lift left angle-of-attack vane until pilot's angle-of-attack indicator needle is centered at the intersection of the green and yellow bands. The shaker shall actuate and L STALL warning light will begin to flash. If necessary, adjust left Rx potentiometer until shaker and warning light actuation occurs at this point.
(10) Slowly lower angle-of-attack vane. The shaker will stop and warning light will extinguish.
(11) Slowly raise angle-of-attack vane and assure that shaker and warning light energize as angle-of-attack indicator needle meets the intersection of the green and yellow bands.
(12) Extend flaps to full down.
(13) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ●●●● . The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line. If necessary, adjust the left pusher 40° potentiometer until pusher actuates at this point.
(14) Return the left vane to the down position and then slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line labeled ●●●● . The shaker shall actuate and the L STALL warning light will flash and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands. If necessary, adjust the left shaker 40° potentiometer until shaker and warning light actuates at this point.
(15) Retract flaps to 20°.
(16) Slowly lift left angle-of-attack vane until aft upper edge of vane is centered on upper line marked ●●● . Pusher shall actuate and pilot's angle-of-attack indicator shall be at the red line. If necessary, adjust left pusher 20° potentiometer until pusher actuates at this point.
(17) Return the left vane to the down position and then slowly lift left angle-of-attack vane until aft lower edge of vane is centered on the lower line marked ●●● . The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands. If necessary, adjust left shaker 20° potentiometer until shaker and warning light actuates at this point.
(18) Retract flaps to 8°.

(19) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ●● . The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line. If necessary, adjust left pusher 8° potentiometer until pusher actuates at this point.

(20) Return the left vane to the down position and then slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line marked ●● . The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands. If necessary, adjust left shaker 8° potentiometer until shaker and warning light actuates at this point.

(21) Retract flaps to full up (assure flaps are all the way up).

(22) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ● . The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line. If necessary, adjust left pusher 0° potentiometer until pusher actuates at this point.

(23) Return the left vane to the down position and then slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on lower line marked ● . The shaker and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands. If necessary, adjust shaker 0° potentiometer until shaker actuates at this point.

I. **Functional Test of Stall Warning System** (Aircraft 25-337, 25-342 and Subsequent, 25-206 thru 25-341 equipped with AMK 81-7, "Horizontal Stabilizer Trim and Autopilot Improvement, 25-061, 25-070 thru 25-205 equipped with AAK 76-4, "Reduced Approach Speed System", and AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement.")

NOTE: This functional test must be performed to comply with the current inspection interval specified in Chapter 5.

(1) Connect electrical power to aircraft.

CAUTION: AVOID CONTACT WITH HOT VANES.

(2) Set Pitot Heat Switches on and verify that vanes are heated. Set Pitot Heat Switches off.

(3) With stall vanes completely lowered, set Stall Warning Switches on. Allow 15-minutes warm-up time for stall warning computer. Verify that the following occurs:
 (a) L STALL and R STALL annunciators shall extinguish.
 (b) Pilot's and copilot's angle-of-attack indicator needle shall move into the green area.
 (c) Neither shaker nor pusher shall actuate, even transiently (a slight tremor at the control wheel is acceptable).

(4) Set Stall Warning Switches off. L STALL and R STALL annunciators shall illuminate.

(5) Actuate both squat switches to the air mode (aircraft on jacks) and set Stall Warning Switches on. Verify flaps at 0°.
(6) Set Stall Warning Test Switch (on pedestal switch panel) to L STALL. No response shall be noted.
(7) Set Stall Warning Test Switch (on pedestal switch panel) to R STALL. No response shall be noted.
(8) Actuate squat switches to ground mode. Set Stall Warning Test Switch to L STALL. Verify that the following occurs:
 (a) Pilot's angle-of-attack indicator needle shall start moving from green area toward the red area at a slow rate.
 (b) As the indicator needle moves into the yellow area, the L STALL annunciator shall flash the nudger, and the LH stick shaker shall actuate (there may be a slight lag as the shaker motor comes up to speed).
 (c) As the indicator needle reaches the red line, the L STALL annunciator flashing shall become steady momentarily, pusher shall actuate briefly and automatically stop as the indicator needle falls back into the yellow area. Set Stall Warning Test Switch off and verify that indicator needle falls back into the green area.
(9) Repeat steps (8)(a) through (8)(c) for the RH stall warning system with the Stall Warning Test Switch set to R STALL.
(10) Set Stall Warning Switches off.
(11) Remove attaching parts securing accelerometer to its mounting. Hold accelerometer in its normal position. Use caution handling the accelerometer to prevent damage to the wiring.
(12) Set Stall Warning Switches on. Raise LH stall vane until pusher activates. Hold LH stall vane at this point. With pusher operating, slowly rotate accelerometer about its cylindrical axis until pusher force drops to approximately 1/2. Hold accelerometer in this position.
(13) Slowly raise RH stall vane until copilot stick shaker actuates. Verify that pusher force increases and continues to increase as RH vane is raised.
(14) Lower both stall vanes and return accelerometer to its normal position.
(15) Repeat steps (12) and (13) using the RH stall vane at pusher and LH stall vane at shaker.
(16) Set Stall Warning Switches off and install accelerometer in aircraft.
(17) Connect pitot static tester to the pitot and static systems. (Refer to Chapter 34.)
(18) Set Stall Warning Switches on. Ensure that both stall vanes are lowered to position the angle-of-attack indicator needle in the green band. Ensure that flaps are fully retracted.
(19) Slowly apply vacuum to the system and observe the pilot's and copilot's angle-of-attack indicators. At 22,500 (±750) feet both indicators shall abruptly move toward the yellow area.
(20) Slowly release vacuum from system and remove pitot-static tester.

(21) Extend flaps full down.
(22) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line marked ●●●●. The nudger and shaker shall actuate and the L STALL warning light will flash and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(23) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ●●●●. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(24) Retract flaps to 20°.
(25) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line marked ●●●. The nudger, shaker, and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(26) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ●●●. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(27) Retract flaps to 8°.
(28) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on the lower line marked ●●. The nudger, shaker, and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(29) Slowly lift left angle-of-attack vane until the upper edge of vane is centered on the upper line marked ●●. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(30) Retract flaps to full up (assure flaps are all the way up).
(31) Slowly lift left angle-of-attack vane until the aft lower edge of vane is centered on lower line marked ●. The nudger, shaker, and L STALL warning light shall actuate and the pilot's angle-of-attack indicator shall be at the intersection of the green and yellow bands.
(32) Slowly lift left angle-of-attack vane until the aft upper edge of vane is centered on the upper line marked ●. The pusher shall actuate and the pilot's angle-of-attack indicator shall be at the red line.
(33) Repeat steps (21) through (32) using the right stall warning switch, right angle-of-attack vane, copilot's angle-of-attack indicator, and R STALL warning light.
(34) If the stall warning system complies with this functional test, perform step (36) and return the aircraft to service.
(35) If the stall warning system fails to comply with this functional test, proceed as follows:
 (a) Perform the "Stall Warning System Ground Adjustment" per paragraph 1.J. and the "Stall Warning System Flight Adjustment/Test" per paragraph 1.L. or,
 (b) Perform the Stall Warning System Ground Calibration" per paragraph 1.M.
(36) Set Battery Switches and Stall Warning Switches to off. Install end cover on computer. Install nose compartment doors. Remove aircraft from jacks.

J. Stall Warning System Ground Adjustment (See figure 203.)

NOTE: This ground adjustment procedure is applicable to the following:
a. Aircraft 25-337, 25-342 and Subsequent.
b. Aircraft 25-206 thru 25-341, except 25-337 when modified per AMK 81-7, "Horizontal Stabilizer Trim and Autopilot Improvement."
c. Aircraft 25-061, 25-070 thru 25-205 when modified per AAK 76-4, "Reduced Approach Speed System" and AMK 81-9, "Horizontal Stabilizer Trim and Autopilot Improvement."

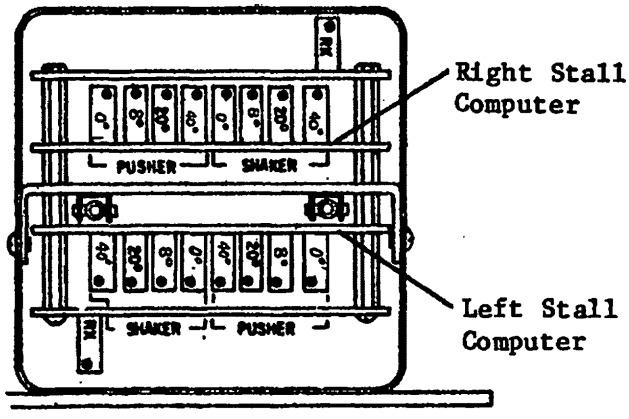
CAUTION: IF ANY COMPONENTS OR AERODYNAMIC SURFACES AFFECTING AIRCRAFT STALL CHARACTERISTICS ARE REPLACED, MODIFIED, OR ADJUSTED, A STALL WARNING SYSTEM GROUND ADJUSTMENT AND A FLIGHT ADJUSTMENT ARE MANDATORY.

(1) Set Battery switches to OFF.
(2) Disconnect auxiliary power if applied.
(3) Remove nose compartment doors.
(4) Connect stall warning computer electrical connectors to aircraft wiring bundle. Remove computer end cover to gain access to potentiometer.
(5) Connect 28 vdc power to aircraft. Set Battery Switches and LH Stall Warning Switch to ON. Allow 15-minutes warm-up time for stall warning computer.
(6) Retract flaps 0°. Ensure that both angle-of-attack vanes are down.
(7) Slowly lift left angle-of-attack vane until pilot's angle-of-attack indicator needle is centered at the intersection of the green and yellow bands. The shaker shall actuate and L STALL warning light will begin to flash. If necessary, adjust left Rx potentiometer until shaker and warning light actuation occurs at this point.
(8) Slowly lower angle-of-attack vane. The shaker will stop and warning light will extinguish.
(9) Slowly raise angle-of-attack vane and assure that shaker and warning light energize as angle-of-attack indicator needle meets the intersection of the green and yellow bands.
(10) Extend flaps to full down 40°.
(11) Slowly lift left angle-of-attack vane until aft upper edge is centered on upper line marked ●●●● . Pusher shall actuate and pilot's angle-of-attack indicator shall be at red line. If necessary, adjust left PUSHER DOWN potentiometer until pusher actuates at this point.
(12) Slowly lift left angle-of-attack vane until aft lower edge is centered on lower line marked ●●●● . Shaker and nudger shall actuate and L STALL warning light shall flash. Pilot's angle-of-attack indicator needle shall be at intersection of green and yellow bands. Adjust left SHAKER DOWN potentiometer until shaker, nudger, and warning light actuate at this point.
(13) Retract flaps to 20°.

(14) Slowly lift left angle-of-attack vane until aft upper edge is centered on upper line marked ●●● . Pusher shall actuate and pilot's angle-of-attack indicator shall be at red line. If necessary, adjust left PUSHER 20° potentiometer until pusher actuates at this point.
(15) Slowly lift left angle-of-attack vane until aft lower edge is centered on lower line marked ●●● . Shaker, nudger, and L STALL warning light shall actuate and pilot's angle-of-attack indicator needle shall be at intersection of green and yellow bands. If necessary, adjust left SHAKER 20° potentiometer until shaker, nudger, and warning light actuate at this point.
(16) Retract flaps to 8°.
(17) Slowly lift left angle-of-attack vane until aft upper edge is centered on upper line marked ●● . Pusher shall actuate and pilot's angle-of-attack indicator needle shall be at red line. If necessary, adjust left PUSHER 8° potentiometer until pusher actuates at this point.
(18) Slowly lift left angle-of-attack vane until aft lower edge is centered on lower line marked ●● . Shaker, nudger, and L STALL warning light shall actuate and pilot's angle-of-attack indicator shall be at intersection of green and yellow bands. If necessary, adjust left SHAKER 8° potentiometer until shaker, nudger, and warning light actuate at this point.
(19) Retract flaps to full up 0°.
(20) Slowly lift left angle-of-attack vane until aft upper edge is centered on upper line marked ● . Pusher shall actuate and pilot's angle-of-attack indicator needle shall be at red line. If necessary, adjust left PUSHER CLEAN potentiometer until pusher actuates at this point.
(21) Slowly lift left angle-of-attack vane until aft lower edge is centered on lower line marked ● . Shaker, nudger, and L STALL warning light shall actuate and pilot's angle-of-attack indicator needle shall be at intersection of green and yellow bands. If necessary, adjust left SHAKER CLEAN potentiometer until shaker, nudger, and warning light actuate at this point.
(22) Repeat steps (5) through (21) using right Stall Warning Switch, right angle-of-attack vane, copilot's angle-of-attack indicator, right stall warning potentiometers, and R STALL warning light.
(23) Set Battery and Stall Warning Switches to OFF.
(24) Disconnect auxiliary power.
(25) Install computer end cover and nose compartment access doors.

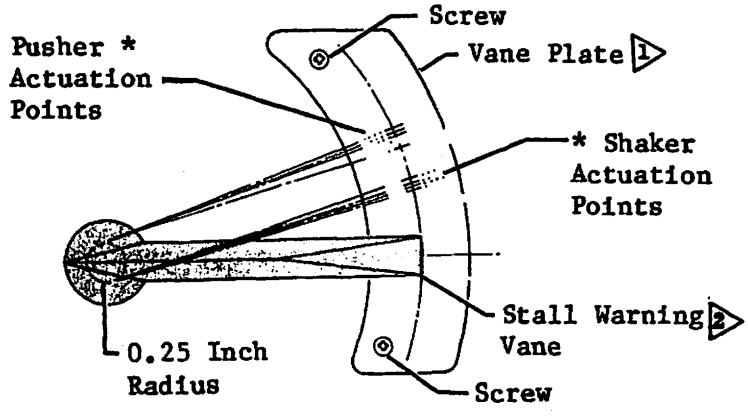
1 Vane plate is bonded to aircraft skin with EC-2216 adhesive, mfd. by 3M Co.

2 Upper travel limits at least 0.25 inch aboved uppermost pusher activation point.



END OPPOSITE PLUG SHOWN
COVER REMOVED

* This is a representation only. Actual and relative mark locations will vary.



Stall Warning Ground Adjustment
Figure 203

K. **Stall Warning System Flight Adjustment** (Aircraft 25-206 thru 25-289 and prior aircraft equipped with AAK 76-4, "Reduced Approach Speed System.")

CAUTION: ° STALL WARNING SYSTEM FLIGHT TESTING MUST BE PERFORMED BY A QUALIFIED PILOT APPROVED BY GATES LEARJET CORPORATION.

° ADJUST C.G. FOR STALL WARNING FLIGHT TEST TO FORWARD (FORWARD LINE TO 3% AFT OF FORWARD).

° FLY STALLS AT 12,000 TO 15,000 FEET (5000 FOOT AGL MINIMUM).

NOTE: ° Make pusher adjustment prior to shaker adjustment. Adjustment of pusher circuitry will affect shaker setting.

° Stall warning light and stick shaker activation are simultaneous. At this point shaker frequency is very low, requiring close attention to detect.

° Stall warning indicator needles out of synchronization at high speeds reveal shaker or pusher actuation speeds are not coordinated between the pilot's and copilot's systems. Readjust 0° potentiometers to correct.

(1) Visually inspect the following systems and components for security of mounting and general condition.
 (a) Flaps, spoilers, and ailerons.
 (b) Wheel well doors.
 (c) Tip tank strakes.
 (d) Wing leading edge (inspect particularly for dents).
 (e) Pitot static ports for condition.

(2) Check flap system for proper rigging. Also check that flap position indicator is properly calibrated. (Refer to 27-50-01.)

NOTE: Exert sufficient upward load on the flaps by hand so as to remove any possible free play when checking flap positions.

(3) Check the spoiler and aileron system rigging.

(4) Check the pilot and copilot airspeed indicators as follows:
 (a) Perform Pitot System Leakage Check using pitot static test set 1811F (manufactured by Barfield Instrument Co.) or equivalent. Assure that leakage rate is not exceeded. Connect test set to both pitot tubes so that both the pilot and copilot airspeed indicators can be checked.

NOTE: Start with a higher pitot pressure than each target pressure. Reduce the pressure until each target pressure is reached.

(b) Adjust the pitot and static tester so that the tester airspeed indicates above 108 knots.
(c) Adjust tester to target airspeed 105 (±3) knots. Record pilot and copilot airspeed readings.
(d) Adjust tester to 95 (±3) knots and 85 (±3) knots and record each pilot and copilot airspeed reading.
(e) Refer to tester airspeed indicator and pilot and copilot airspeed indicator correction cards. Pilot and copilot correct airspeed indication shall be within ±2 knots of corrected airspeed indication at each target setting.

(5) Perform stall warning preflight weight and balance adjustments as follows:
(a) Fill the wing and tip fuel tanks.
(b) Weigh the aircraft. Add ballast as required so that the C.G. is at the forward limit (forward to 3% aft of forward) as shown on the Stall Speeds Charts and the Airplane Flight Manual C.G. chart.
(6) Assure that the Stall Warning Switches are set to OFF.
(7) Remove plug and plate from patch cable access hole. The plug is located on frame 5 at RBL 4, WL 22 or 5 inches below outflow valve.
(8) Remove stall warning computer from nose compartment and place it in the cabin.
(9) Install patch cable connectors into frame 5 bulkhead and connect to ship's wiring and computer. Patch cable P/N 15-1013 (see figure 204) is available from Learjet Spares Department.
(10) Secure nose compartment access doors.
(11) Remove computer end cover.
(12) Perform "Functional Test of Stall Warning System," paragraph 1.G. (steps 21 through 33), to prove integrity of patch cable.
(13) Record aircraft takeoff weight. Approach stall speed by decreasing airspeed at one (1) knot per second. Check each system separately. Fly the aircraft in each of the following configurations:
Gear Down - Flaps Down (40°)
Gear Up - Flaps 20°
Gear Up - Flaps 8°
Gear Up - Flaps up (0°)
(a) Pusher actuation speed shall fall on stall speed curve ±2 knots for each configuration. Refer to Approved Flight Manual (Section IV) for stall speeds. Enter stall speed charts using instrument-corrected airspeed as provided by correction card for Pilot's Airspeed Indicator. Pusher must actuate before aerodynamic stall.
(b) Demonstrate that at least a 3 knot margin exists between pusher airspeed and aerodynamic stall.
(c) Shaker shall actuate at least 10% above the pusher speed in 8°, 20°, and 40° flap settings and 7% at 0° flap setting.
(d) Angle-of-attack needle shall enter yellow band at shaker and red band at pusher.

(e) Adjust system as required until requirements of steps (13)(a), (b), (c), and (d) are met.

NOTE: Roll characteristics shall remain acceptable to pusher speed (no roll-off beyond 20° bank).

(f) Recheck system. Pusher speed shall be within ±2 knots of speeds reflected in the Approved Airplane Flight manual stall speed charts.

(14) After pusher-shaker has been adjusted, approach a stall at a rate greater than 1 knot per second. Angle of attack rate feature shall actuate pusher before aerodynamic stall.

(15) Climb to 22,500 (±750) feet.

(a) Pilot and copilot angle-of-attack indicator needles shall abruptly move toward yellow band, indicating correct altitude switch function.

(16) Remove patch cable and install stall warning computer in nose avionics compartment. Assure that plugs are bottomed in receptacles. Secure attaching parts.

(17) Replace frame 5 access hole plug and plate. Seal pressure side with ProSeal 890B.

CAUTION: DO NOT PERFORM ANY ADJUSTMENTS OF COMPUTER PUSHER AND SHAKER ACTUATION POINTS AFTER COMPLETION OF THE FLIGHT CALIBRATION.

(18) Repeat "Functional Test of Stall Warning System," paragraph 1.G. (steps 21 through 33), to verify that the existing vane plate markings for pusher and shaker match the flight calibrated computer. If vane plate markings match computer, return aircraft to service. If vane plate markings DO NOT match computer, then a new vane plate must be installed and marked to match computer pusher and shaker actuation points.

L. **Stall Warning System Flight Test/Adjustment** (Aircraft 25-290 and Subsequent and prior aircraft equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers.")

CAUTION: STALL WARNING SYSTEM FLIGHT TEST/ADJUSTMENT MUST BE PERFORMED BY A PILOT APPROVED BY GATES LEARJET CORPORATION FOR STALL WARNING CALIBRATION.

NOTE: ° Make pusher adjustments prior to shaker adjustment. Adjustment of pusher circuit will affect shaker setting.

° Stall warning light and stick shaker activation are simultaneous.

° Stall warning indicator needles which are out of synchronization at high speeds reveal that shaker or pusher actuation speeds are not coordinated between the pilot's and copilot's systems. Readjust CLEAN potentiometers to correct.

(1) Visually inspect the following systems and components for security of mounting and general condition.
 (a) Flaps, spoilers, and ailerons.
 (b) Wheel well doors.
 (c) Tip tank strakes.
 (d) Wing leading edge (inspect particularly for dents).
 (e) Pitot static ports for condition.

(2) Check flap system for proper rigging. (Refer to 27-50-01.)

NOTE: Exert sufficient upward load on the flaps by hand as to remove any possible free play when checking flap positions.

(3) Check spoiler and aileron system rigging.

(4) Check pilot and copilot airspeed indicators as follows:
 (a) Perform Pitot System Leakage Check using pitot-static test set 1811F (manufactured by Barfield Instrument Co.) or equivalent. Assure that leakage rate is not exceeded. Connect test set to both pitot tubes so that both the pilot and copilot airspeed indicators can be checked.

NOTE: Start with a higher pitot pressure that each target pressure. Reduce pressure until each target pressure is reached.

 (b) Adjust pitot-static tester so that tester airspeed indicator indicates above 108 knots.
 (c) Adjust the tester to the target airspeed 105 (±3) knots. Record pilot and copilot airspeed ratings.
 (d) Adjust tester to 95 (±3) knots and 85 (±3) knots and record each pilot and copilot airspeed reading.

(e) Refer to tester airspeed indicator and pilot and copilot airspeed indicator correction cards. Pilot and copilot corrected airspeed indication shall bewithin ±2 knots of corrected airspeed indication at each target setting.

(5) Perform stall warning preflight weight and balance adjustments as follows:
(a) Fill the wing and tip fuel tanks.
(b) Weigh the aircraft. Add ballast as required so that the C.G. is at the forward limit (forward to 1.54% aft of forward) as reflected in the Approved Airplane Flight Manual C.G. chart and stall speed charts.

(6) Assure that Stall Warning Switches are set to OFF.

(7) Remove plug and plate from patch cable access hole on frame 5 at RBL 4, WL 22, or 5 inches below outflow valve.

(8) Remove stall warning computer from nose compartment and place it in the cabin.

(9) Install patch cable connectors into frame 5 bulkhead and connect to ship's wiring and computer. Patch cable P/N 15-1013 (see figure 204) is available from Learjet Spares Department.

(10) Secure nose compartment access doors.

(11) Remove computer end cover.

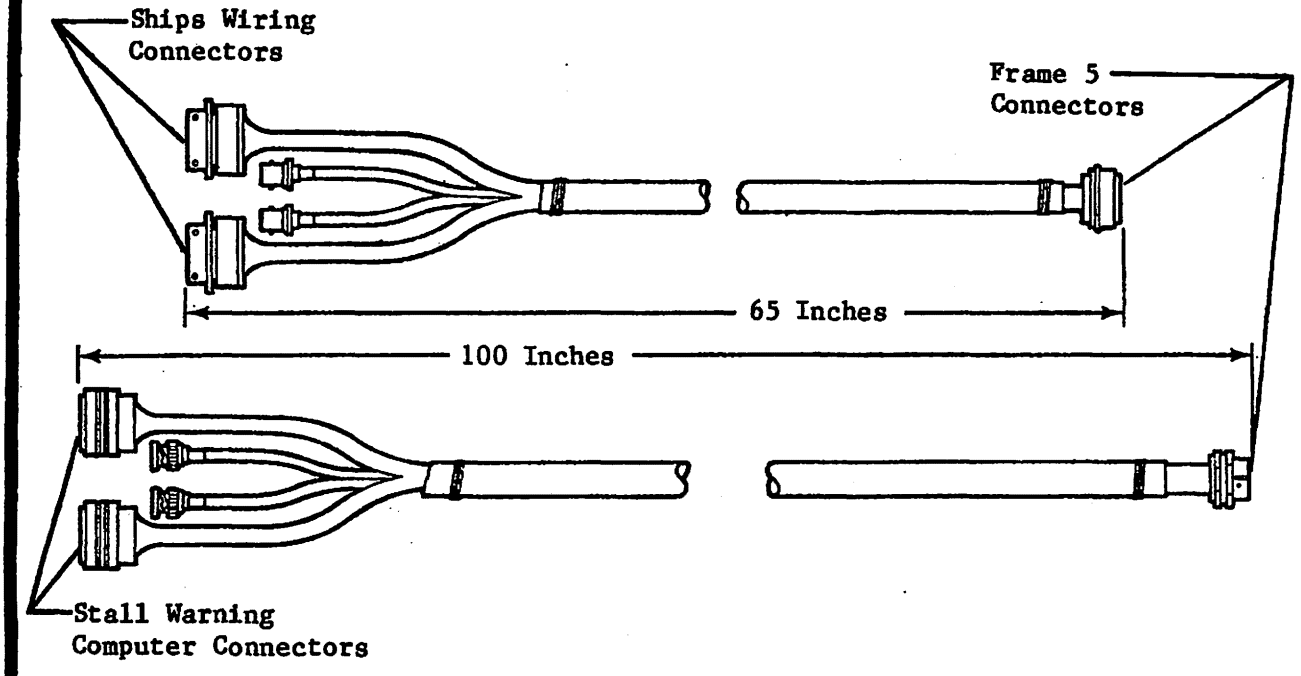
(12) Perform "Functional Test of Stall Warning System," paragraph 1.G. (steps 21 through 33), to prove integrity of patch cable.

(13) Flight test aircraft as follows:
(a) The stall warning computer will be calibrated with the aircraft configured as follows:
Gear Up - Flaps Up (0°)
Gear Down - Flaps 8°
Gear Down - Flaps 20°
Gear Down - Flaps Down (40°)

NOTE: Flight test aricraft at 8,000 to 15,000 feet (5,000 Foot AGL).

(b) Record aircraft takeoff weight.
(c) Check angle of attack flap biasing at 3°, 13°, and 25°.
(d) Determine Chart Stall Speed for each configuration. Trim to 1.3 times stall speed.

WARNING: DO NOT STALL THE AIRCRAFT AT ANY C.G. AFT OF FORWARD WITH STALL WARNING SWITCHES SET TO OFF (PUSHER OFF).



NOTE: Patch Cable P/N 15-1013 contains both forward and aft patch cables.

Stall Warning System Flight Adjustment Patch Cable
Figure 204

(e) Set Stall Warning Switches to OFF. At approximately 12,000 feet, gradually increase back pressure and fly the aircraft to aerodynamic stall at a deceleration rate of 1 knot per second or less. Note corrected stall speed. Aircraft will remain controllable in roll until aerodynamic stall/aft stick is reached. No roll-off of greater than 20° bank is allowable throughout stall speed recovery.

(f) Record aerodynamic stall/aft stop speed for the configuration being calibrated. Actual aerodynamic stall speed must not exceed by more than 3 knots the stall speed reflected in the Approved Airplane Flight Manual stall speed charts.

(g) Angle-of-attack needle shall enter yellow band at shaker and red band at pusher.

(h) With Stall Warning Switches ON, set stall warning system pusher to actuate at ±3 knots of published stall/pusher speed in each configuraion. Pusher must actuate at or above aerodynamic stall.

(i) Set shaker to actuate a minimum of 7 knots or 7% above pusher speed.

(j) After setting pusher/shaker for each configuration, record pusher/shaker speed and gross weight at that configuration.

NOTE: Roll characteristics shall remain acceptable to pusher speed.

(14) After pusher/shaker has been adjusted, approach the stall at a rate greater than 1 knot per second. The angle of attack rate shall actuate the pusher prior to aerodynamic stall.

(15) Climb to 22,500 (±750) feet.

(a) Pilot and copilot angle-of-attack indicator needles shall abruptly move toward yellow band, indicating correct altitude switch function.

(16) Remove patch cable and install stall warning computer in nose avionics compartment. Assure that plugs are bottomed in receptacle. Secure attaching parts.

(17) Replace frame 5 access hole plug and plate. Seal pressure side with ProSeal 890B.

CAUTION: DO NOT PERFORM ANY ADJUSTMENTS OF COMPUTER PUSHER AND SHAKER ACTUATION POINTS AFTER COMPLETION OF THE FLIGHT CALIBRATION.

(18) Repeat "Functional Test of Stall Warning System," paragraph 1.G. or 1.I., as applicable (steps 21 through 33), to verify that the existing vane pate markings for pusher and shaker match the flight calibrated computer. If the vane plate markings match the computer, return aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to precisely match the computer pusher and shaker actuation points. (Refer to 27-32-10.)

M. **Stall Warning System Ground Calibration** (Aircraft 25-290 and Subsequent and prior aircraft equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers.") (See figure 205.)

WARNING: STALL WARNING SYSTEM GROUND CALIBRATION APPLIES TO STANDARD CONFIGURATION AIRCRAFT. DOES NOT APPLY TO AIRCRAFT EQUIPPED WITH CAMERAS, EXTERNAL STORES, RADOMES, OR PYLONS.

NOTE: ° This calibration procedure shall be performed when a stall vane, transducer, computer, or indicator is replaced or if the stall warning system has failed the Functional Test of Stall Warning System (paragraph 1.G. or 1.I.).

° Stall Warning System Ground Adjustment and Flight Adjustment/Test procedures (paragraph 1.H. or 1.L.) can be used in lieu of these procedures.

(1) Tools and Equipment

NAME	PART NUMBER	MANUFACTURER	USE
Inclinometer	T3500002-600	Gates Learjet	Calibration procedure.
Test Box	5471100-1	Gates Learjet	Calibration procedure.
Mylar Overlay	5471102-5 (LH) 5471102-6 (RH)	Gates Learjet Gates Learjet	

(2) Set Battery Switches and Stall Warning Switches off.
(3) Place aircraft on jacks. Assure that wheels are clear of ground.
(4) Level aircraft in the pitch axis (refer to Chapter 8) or place inclinometer tool (P/N T3500002-600) on cabin seat rail with FWD arrow pointing forward. Assuring that inclinometer is free of obstructions, raise or lower nose jack to center inclinometer bubble.

NOTE: Move inclinometer to other areas of seal rails to verify leveling. A long precision bar (18 to 36 inch) can be used to average uneven area of the seat rails.

(5) Place a piece of tape forward of LH stall vane so that rubber-tipped plunger of inclinometer will not mar painted surface. Install inclinometer (P/N T3500002-600) on LH stall vane. Secure tool finger-tight only and release tension shaft to the extended position.

NOTE: Locate inclinometer on stall vane so that approximately 0.25 inch gap exists between the retracted tension shaft rubber tip and aircraft skin.

(6) Remove nose compartment access doors.
(7) Disconnect electrical connector and coax connectors from stall warning computer.
(8) Connect patch cable (W2, LH side) between test box (P/N 5471100-1), stall warning computer (LH side), and aircraft wiring (P913, LH side).

NOTE: No connections are made to coax cables.

(9) Remove attaching parts and cover (end opposite electrical connectors) from stall warning computer.
(10) Set Battery and L Stall Warning Switches on. Allow 15-minutes warm-up time for stall warning computer. Check that PWR annunciator on test box is illuminated and that SQ SW annunciator is extinguished. Block LH squat switches to the GND mode. Check that SQ SW annunciator is illuminated.
(11) Adjust CAL VANE LEVEL as follows:
 (a) Set CAL Selector Switch to VANE LEVEL.
 (b) Rotate Calibration Flap Selector Knob to CAL.
 (c) Level LH stall vane at 0° using the inclinometer. Release tension shaft to hold stall vane in this position.
 (d) Adjust CAL VANE LEVEL potentiometer to obtain zero (null) reading on null meter.
 (e) Set and hold SENS Switch to NULL. Adjust CAL VANE LEVEL potentiometer to obtain zero (null) reading on null meter. Release SENS Switch.
(12) Adjust CAL VANE HI as follows:
 (a) Set CAL Selector Switch to VANE HI.
 (b) Pull out on inclinometer tension shaft and level LH stall vane at 33-1/3 degrees. Release tension shaft to hold stall vane in this position.
 (c) Adjust CAL VANE HI potentiometer to obtain zero (null) reading on null meter.
 (d) Set and hold SENS Switch to NULL. Adjust CAL VANE HI potentiometer to obtain zero (null) reading on null meter. Release SENS Switch.
(13) Repeat steps (11) and (12) until no further adjustment is required. Null meter shall be 0 (±4) marks with SENS Switch set to NULL.

NOTE: ° Do not disturb these calibration adjustments during the remainder of the test for the LH system.

° Adjust potentiometer very slowly to ensure accurate results.

° Aircraft static system must be lower than 22,500 feet during the following adjustments.

(14) Rotate Calibration Flap Selector Knob to RX.

(15) Rotate digital potentiometer for an indication of 100 full right on test set meter.
(16) Verify that pilot's angle-of-attack indicator needle is at the intersection of green and yellow color bands. Slowly adjust digital potentiometer on test set to move indicator needle to this point if necessary.
(17) Adjust left stall system R_X potentiometer until test box SHAKE and WARN annunciators are just at the threshold of blinking.

NOTE: Do not disturb adjustment R_X potentiometer after completion of step 17.

(18) Rotate Calibration Flap Selector Knob to 0°. Rotate digital potentiometer to 0° pusher values. (Refer to table I, figure 205.)

NOTE: Adjust pusher potentiometer first, then shaker. If any pusher potentiometer is readjusted, then the corresponding shaker potentiometer must also be readjusted.

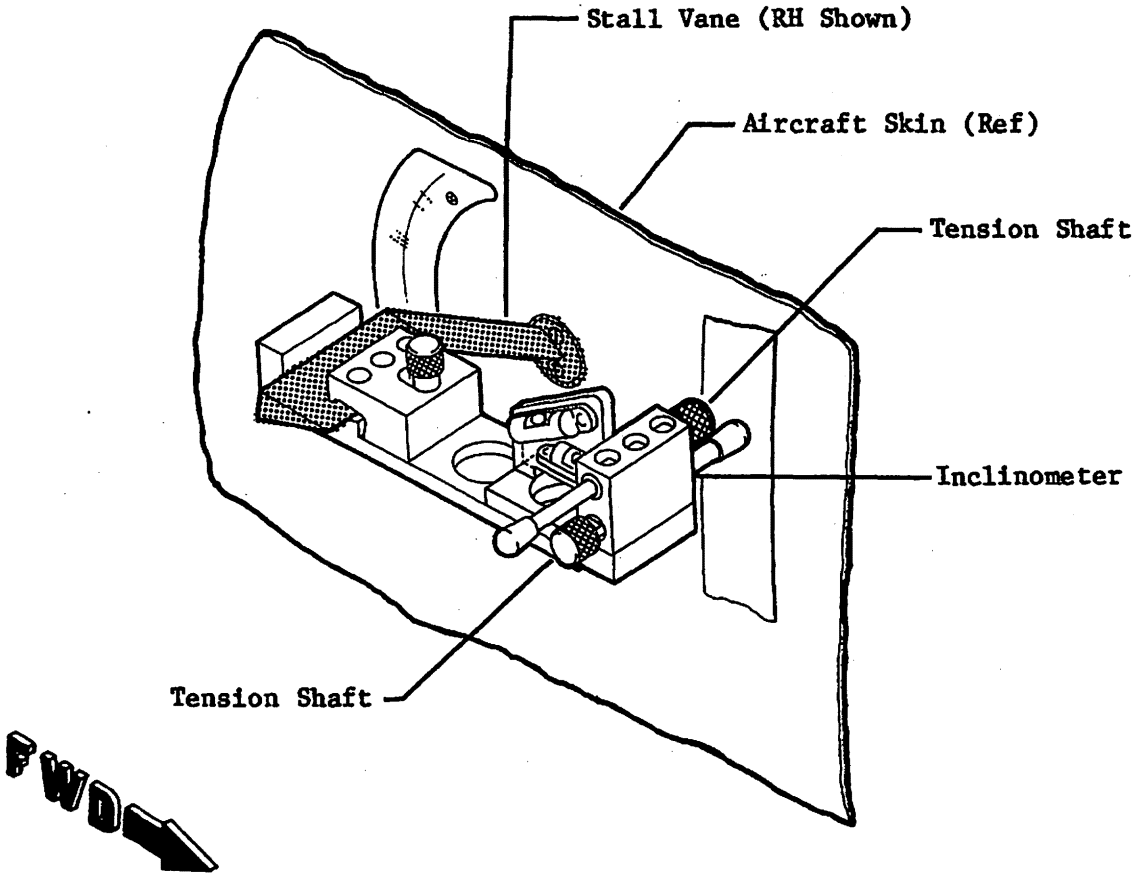
(19) Adjust left system 0° PUSHER potentiometer for threshold illumination of test box PUSH annunciator. Null meter shall read approximately zero.
(20) Rotate digital potentiometer below 0° pusher value. Threshold illumination must occur within ±2 units of the clean pusher value as the digital potentiometer setting is increased. Repeat steps (19) and (20) if necessary.
(21) Rotate digital potentiometer to 0° shaker values. (Refer to table 1, figure 205.)
(22) Adjust left stall system 0° SHAKER potentiometer for threshold blinking of the test box SHAKE and WARN annunciators. Null meter shall read approximately 40 right.
(23) Rotate digital potentiometer below the 0° shaker value. Threshold blinking shall occur within ±2 units of 0° shaker value.
(24) Repeat steps (18) thru (23) using the Calibration Flap Selector Knob at the remaining flap settings (8°, 20°, and 40°).
(25) Using values in table I, figure 205, rotate digital potentiometer to each value at each flap setting and verify that SHAKE, WARN, and PUSH annunciators blink within ±2 units. If necessary, adjust potentiometers.

NOTE: Changing any pusher adjustment requires that the corresponding shaker potentiometer also be adjusted.

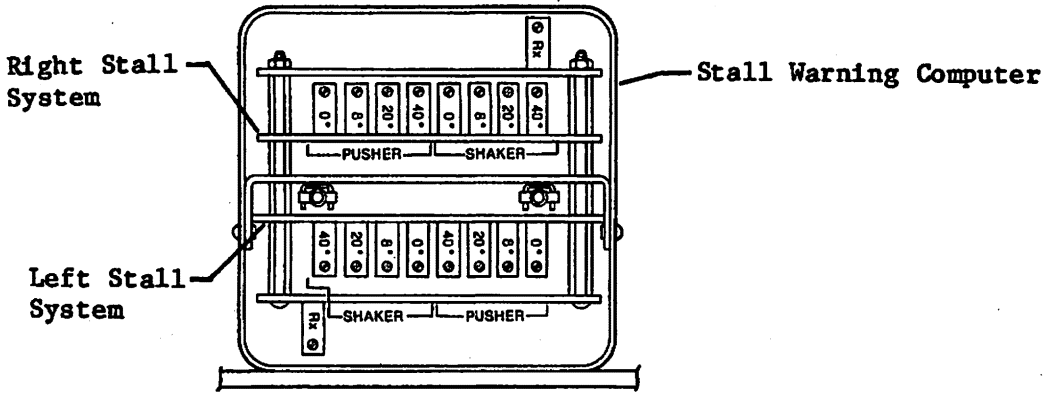
(26) Rotate Calibration Flap Selector knob to 0°.
(27) Rotate digital potentiometer to 108 and verify that pilot's angle-of-attack indicator needle is at 5th mark (± one needle width) from green and yellow intersection in green area. Adjust left stall system 0° SHAKER potentiometer if necessary.

Calibration Flap Selector Switch Setting	Digital Pot Setting	
	Pusher ±2	Shaker ±2
0°	750	555
8°	900	585
20°	750	480
40°	690	435

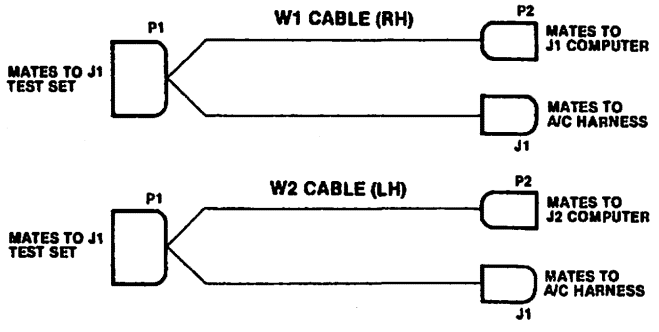
Table I



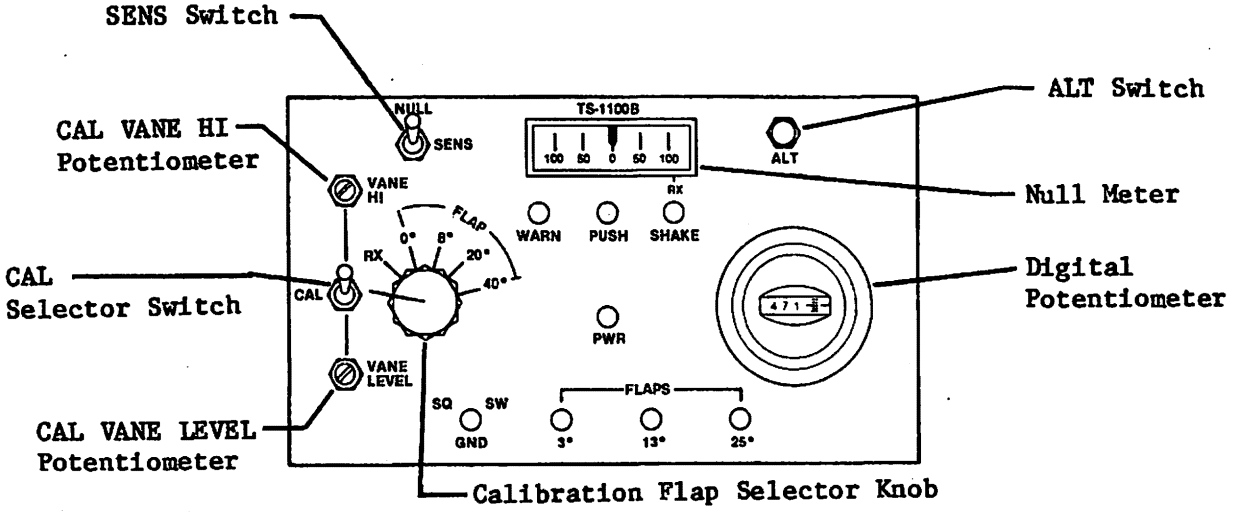
Stall Warning System Ground Calibration
Figure 205 (Sheet 1 of 2)



END OPPOSITE PLUG SHOWN
COVER REMOVED



Patch Cable



14-172B
9-343A

Stall Warning System Ground Calibration
Figure 205 (Sheet 2 of 2)

(28) Rotate digital potentiometer to a meter indication of 50 on right side of zero. Depress and hold ALT Switch and verify null meter deflection to the left.
(29) Cycle flaps to 0°, 8°, 20°, and 40° positions. As flaps pass through 3°, 13°, and 25° the respective FLAPS annunciators on test box shall illuminate.
(30) Remove inclinometer from LH stall vane and install on RH stall vane.
(31) Set Battery Switches and Stall Warning Switches off.
(32) Disconnect patch cable (W2, LH side) from test box, computer and aircraft wiring.
(33) Connect patch cable (W1, RH side) between test box, computer (RH side) and aircraft wiring (P914, RH side). Repeat steps M.(10) through M.(29) for the RH stall warning system.
(34) Set Battery Switches and Stall Warning Switches off. Position cover on end of stall warning computer and secure with attaching parts.
(35) Disconnect patch cable (W1, RH side) from test box, computer, and aircraft wiring. Connect electrical connectors and coax cables to computer.
(36) Perform Functional Test of Stall System, paragraph 1.G., or 1.I., as applicable (steps 21 through 33), to verify that the existing vane plate markings for pusher and shaker match the calibrated computer. If the vane plate markings match the computer, return the aircraft to service. If the vane plate markings DO NOT match the computer, then a new vane plate must be installed and marked to precisely match the computer pusher and shaker actuation points. (Refer to 27-31-10.)

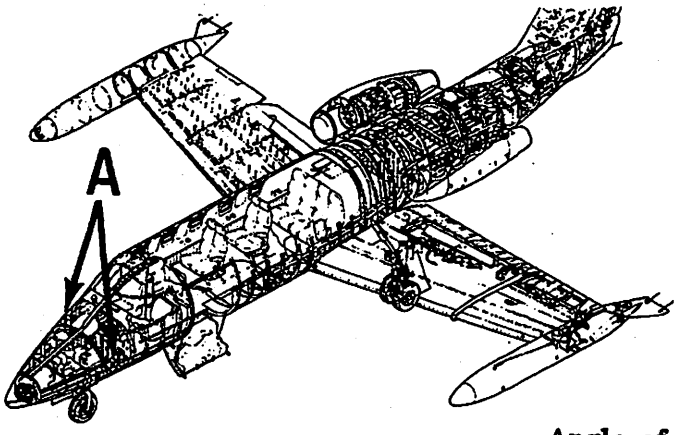
ANGLE-OF-ATTACK TRANSDUCER - MAINTENANCE PRACTICES

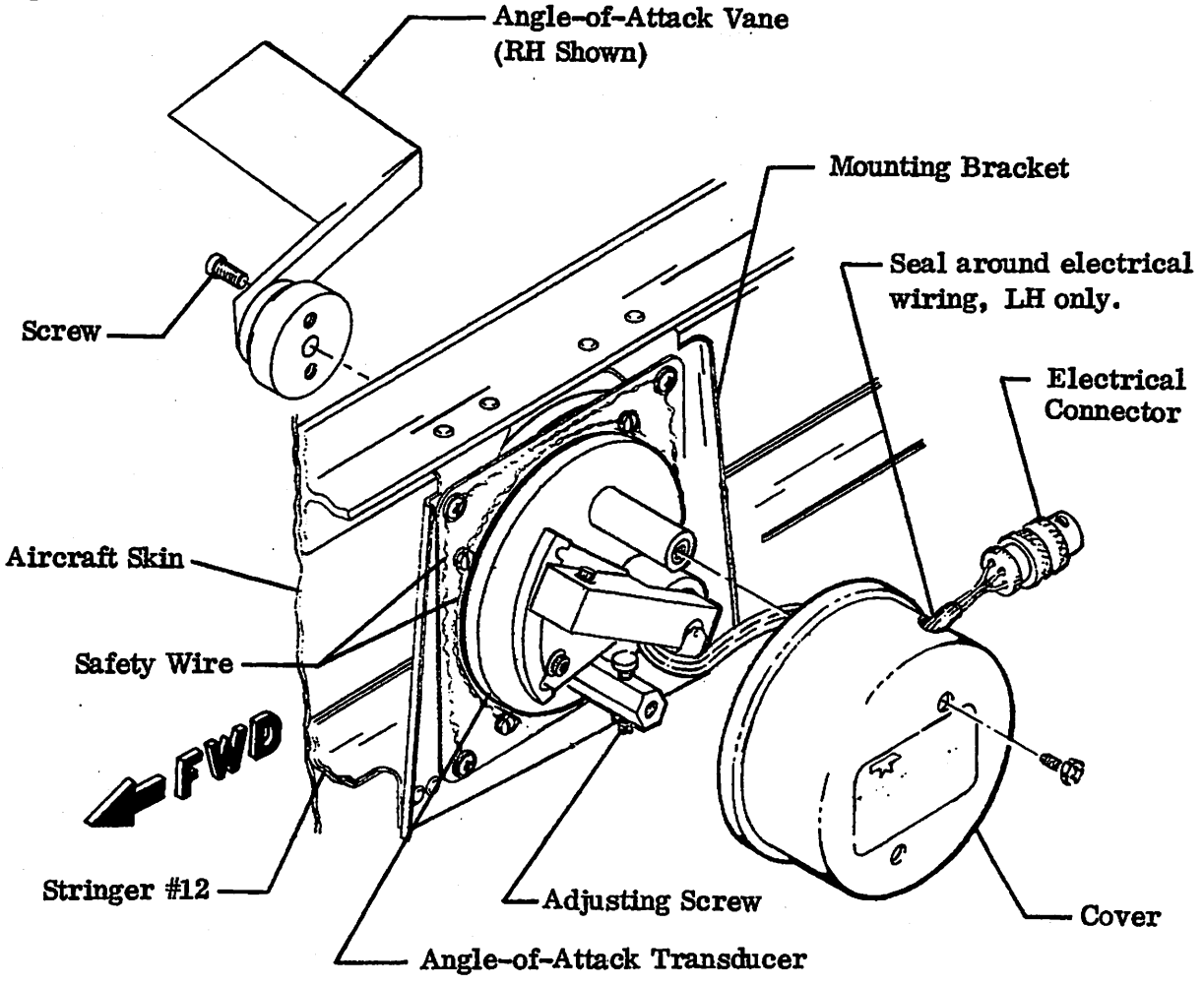
1. Removal/Installation

NOTE: Removal and installation procedures are the same for both angle-of-attack transducers.

A. Remove Angle-of-Attack Transducer (See figure 201)
 (1) Remove nose compartment access doors.
 (2) Remove screws securing vane to transducer. Remove vane from transducer.
 (3) Remove electrical power from aircraft.
 (4) Disconnect electrical connector from transducer.
 (5) Remove safety wire, attaching parts, and mounting bracket with transducer attached from aircraft.

B. Install Angle-of-Attack Transducer (See figure 201)
 (1) Install transducer; secure with attaching parts and safety wire.
 (2) Connect electrical connector to transducer.
 (3) Install vane on transducer and secure with attaching parts.
 (4) Install nose compartment access doors.
 (5) Restore electrical power to aircraft.





RH Transducer Shown

Detail A

Angle-of-Attack Transducer Installation
Figure 201

8-23C-1

CONTROL COLUMN SHAKER - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Control Column Shaker (See figure 201)

(1) Set Battery Switch(es) and Stall Warning Switch(es) to OFF.

(2) Disconnect electrical wiring from control column connector.

(3) Remove attaching parts and shaker from control column.

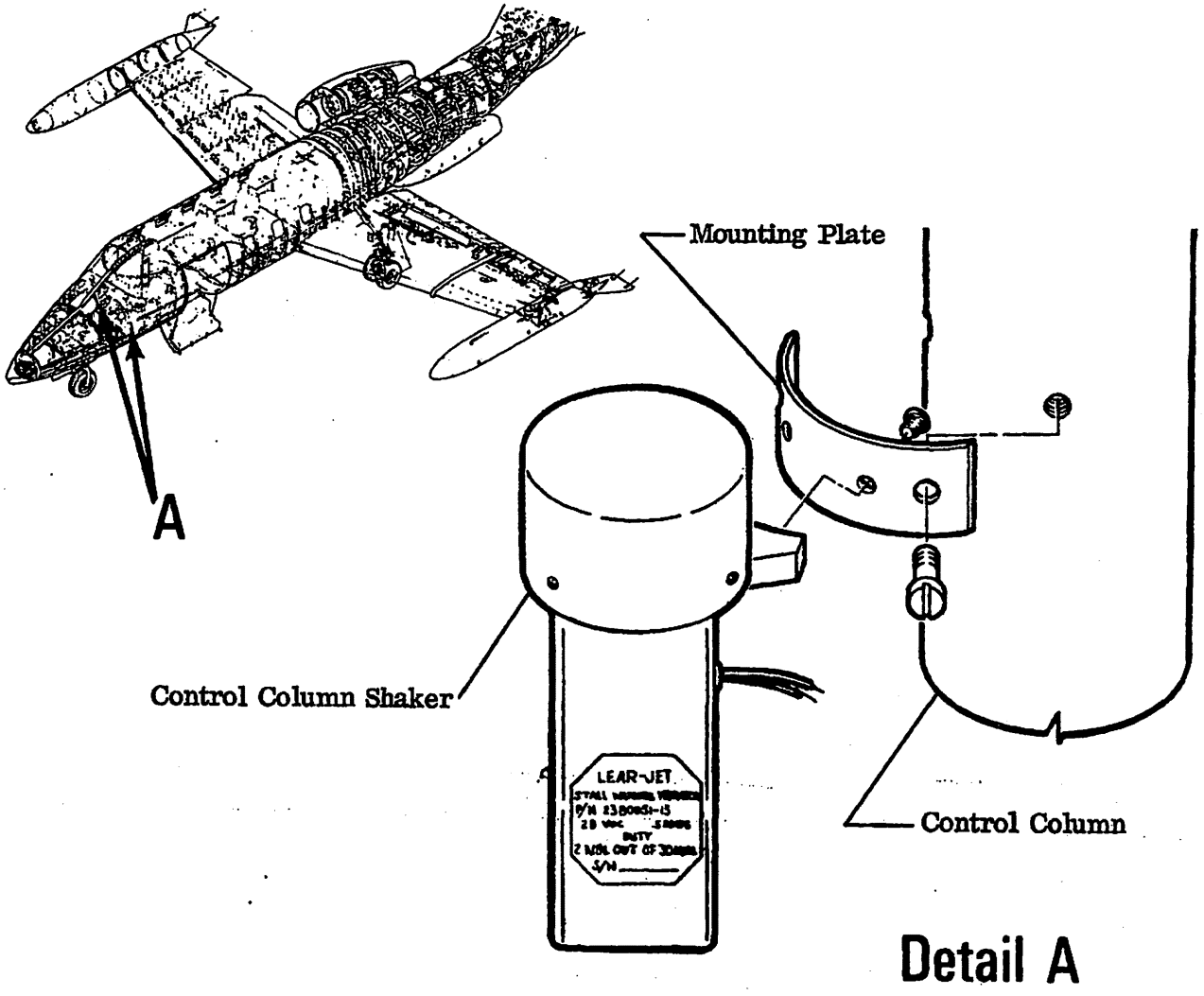
B. Install Control Column Shaker (See figure 201)

(1) Install shaker on control column and secure with attaching parts.

(2) Install shrink tubing (P/N RNF 100 type T), over wiring.

(3) Crimp new pins (P/N 66107-1) to wires and insert wires into electrical connector.

(4) Secure wiring to control column clear of interference with surrounding structures.



Control Column Shaker Installation
Figure 201

2. Adjustment/Test

A. Functional Test of Control Column Shaker (See figure 202)

NOTE: • This functional test is to be performed in accordance with current inspection intervals in Chapter 5.

• The purpose of this functional test is to verify the control column shaker performance by measuring frequency.

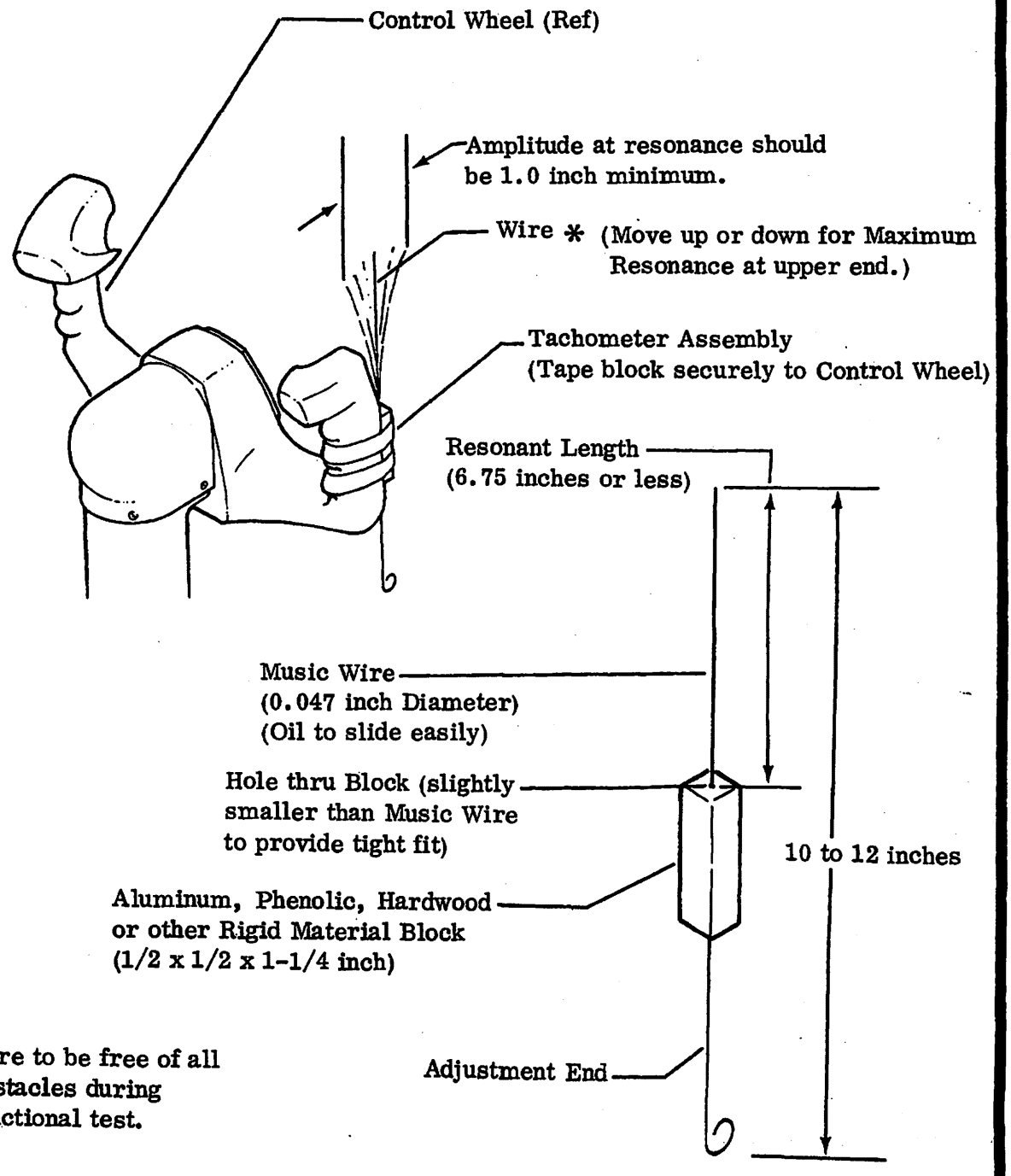
(1) Fabricate a tachometer assembly as shown in figure 202.
(2) Check shaker mounting screws for tightness. Looseness will "unload" the shaker, change the frequency, and reduce the force transmitted.
(3) Trim control column to mid-travel so that elevator stops are not contacted and little or no manual force is required to hold the column upright during this procedure.
(4) Tape tachometer to LH control wheel as shown in figure 202 so that straight end is free to vibrate.
(5) Set Battery Switch(es) and LH Stall Warning Switch to ON.
(6) Raise LH stall vane far enough to get a steady state shaker (no pulsing), but not far enough to engage pusher.
(7) Holding control wheel firmly with one hand, adjust exposed length of wire for maximum resonance.

NOTE: Resonance is easily recognized when the end of the wire vibrates with an amplitude of one inch or more. Adjust length of wire as required to produce the maximum amplitude of vibration at tip of wire. Ensure that the control column is not on stops or touching anything except operator's hand during this operation, since any such contact could change frequency.

(8) Lower LH stall vane, stopping shaker operation.

CAUTION: DO NOT DISTURB WIRE LENGTH.

(9) Carefully measure resonant length of wire. Wire length should be no longer than 6.75 inches.
(10) If the resonant length of wire is greater than 6.75 inches, there is a problem in the system. Longer resonant length means too low a frequency, which may be caused by a weak motor, low input voltage, or loose mounting screws.
(11) Perform steps A. (2) through A. (10) using the RH control wheel, RH Stall Warning Switch and RH stall vane.



Tachometer Assembly
Figure 202

STALL WARNING COMPUTER-AMPLIFIER - MAINTENANCE PRACTICES

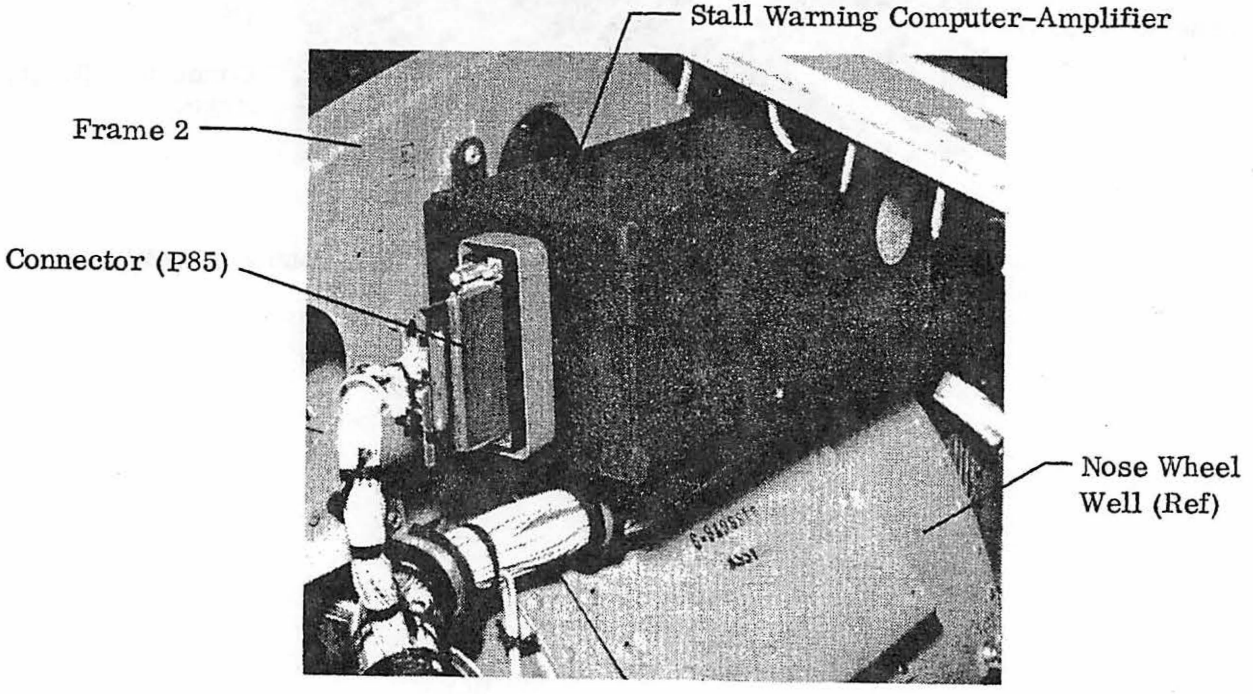
1. Removal/Installation

A. Remove Computer-Amplifier (See figure 201)

(1) Remove nose compartment access doors.
(2) Remove electrical power from aircraft.
(3) Disconnect electrical connectors from computer-amplifier.
(4) Remove attaching parts and computer-amplifier from aircraft.

B. Install Computer-Amplifier (See figure 201)

(1) Install computer-amplifier and secure with attaching parts.
(2) Connect electrical connectors to computer-amplifier.
(3) Install nose compartment access doors.
(4) Restore electrical power to aircraft.
(5) Perform Negative G Accelerometer Functional Test. (Refer to 27-31-06.)
(6) Perform the Stall Warning Functional Tests and make adjustments as necessary. (Refer to Adjustment/Test in 27-31-00.)

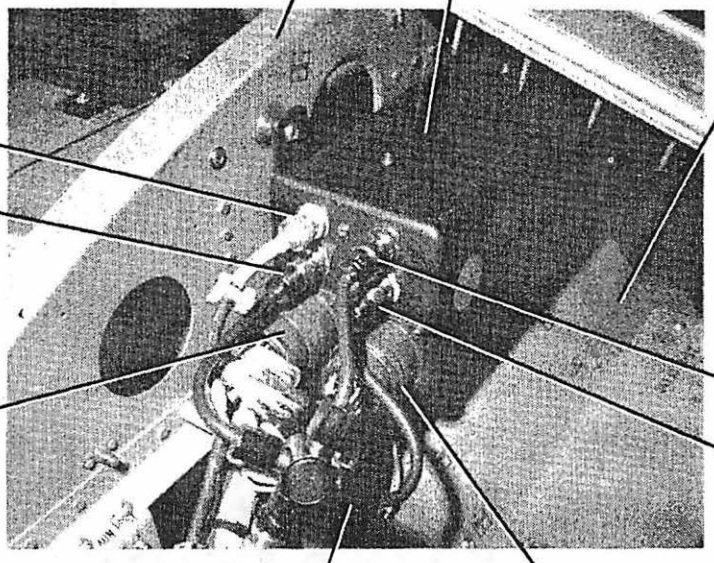


25-061, 25-070 thru 25-205 not Equipped with Reduced Approach Speed System

CAUTION: CONSULT WIRING MANUAL TO ENSURE PROPER CONNECTOR MATING TO COMPUTER-AMPLIFIER RECEPTACLE.

Stall Warning Computer-Amplifier Installation
Figure 201 (Sheet 1 of 2)

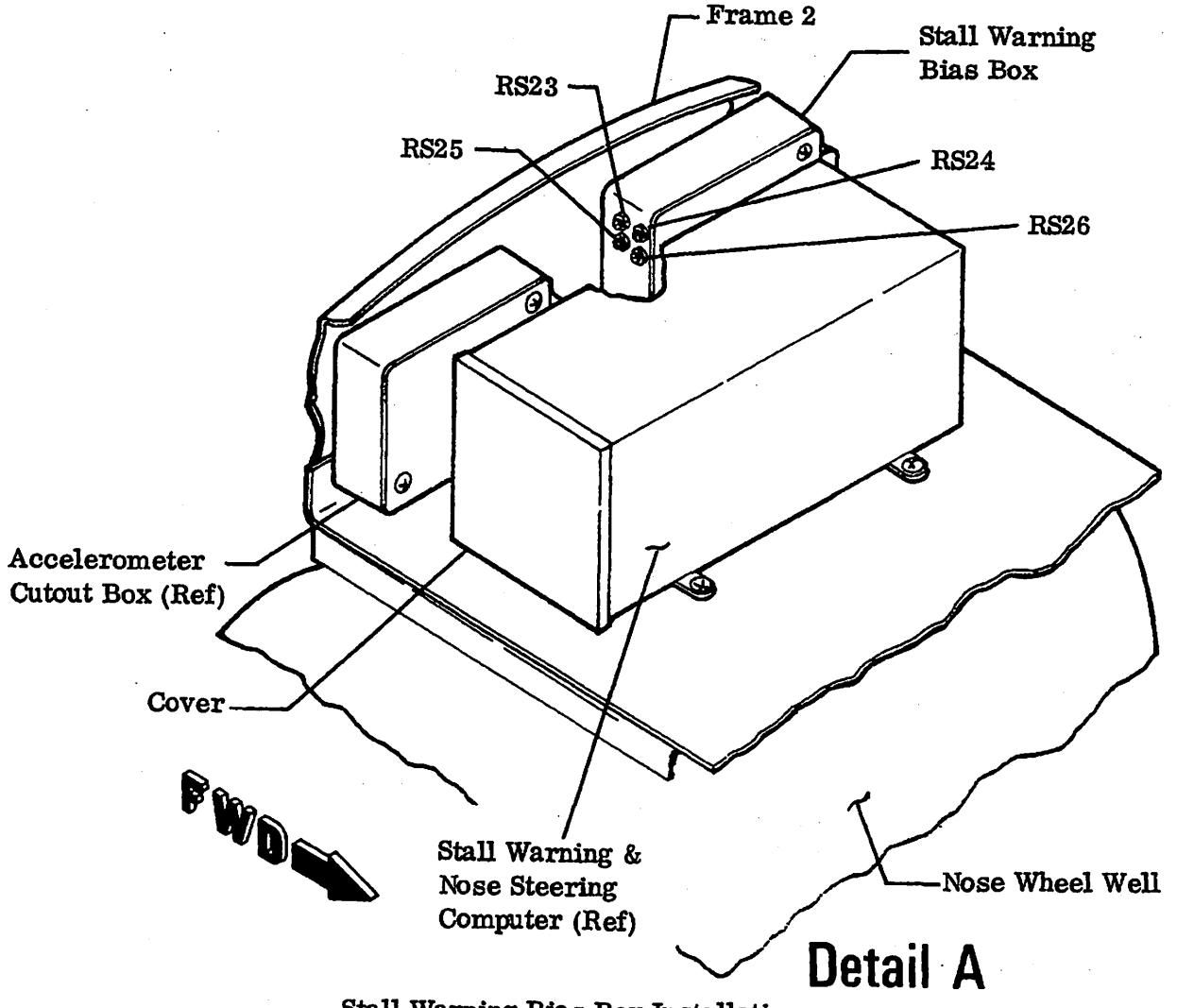
EFFECTIVITY: ALL



Stall Warning Computer-Amplifier Installation
Figure 201 (Sheet 2 of 2)

STALL WARNING BIAS BOX - MAINTENANCE PRACTICES

1. Removal/Installation
 - A. Remove Bias Box (See figure 201)
 - (1) Remove nose compartment access doors.
 - (2) Remove electrical power from aircraft.
 - (3) Disconnect electrical connector from bias box.
 - (4) Remove attaching parts and bias box from aircraft.
 - B. Install Bias Box (See figure 201)
 - (1) Install bias box and secure with attaching parts.
 - (2) Connect electrical connector to boas box.
 - (3) Install nose compartment access doors.
 - (4) Restore electrical power to aircraft as required.



Stall Warning Bias Box Installation
Figure 201

9-1141B

EFFECTIVITY: Aircraft 25-070 thru 25-205 not Equipped with Reduced Approach Speed System

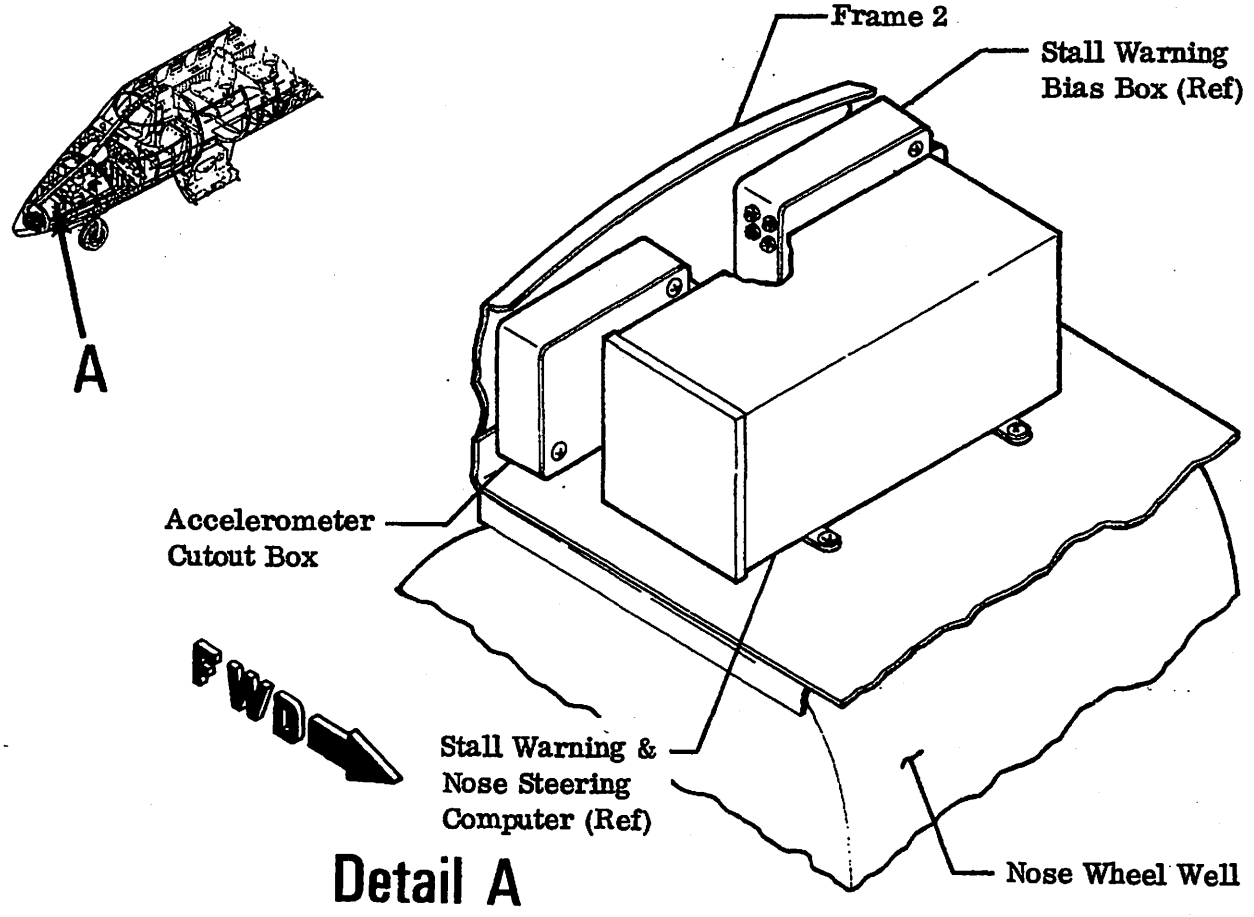
STALL WARNING ACCELEROMETER CUTOUT BOX - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: On aircraft certified for operation in Australia, the accelerometer cutout box is identified as the series pusher control box.

A. Remove Accelerometer Cutout Box (See figure 201)
 (1) Remove nose compartment access doors.
 (2) Remove electrical power from aircraft.
 (3) Disconnect electrical connector from accelerometer cutout box.
 (4) Remove attaching parts and accelerometer cutout box from aircraft.

B. Install Accelerometer Cutout Box (See figure 201)
 (1) Install accelerometer cutout box and secure with attaching parts.
 (2) Connect electrical connector to accelerometer cutout box.
 (3) Install nose compartment access door.
 (4) Restore electrical power to aircraft.



Stall Warning Accelerometer Cutout Box Installation
Figure 201

9-114B

EFFECTIVITY: Aircraft 25-070 thru 25-205 not Equipped with Reduced Approach Speed System

STALL WARNING ACCELEROMETER - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: To assure that the negative "G" accelerometer in the stall warning system is operating properly, the accelerometer must be operationally checked in accordance with current inspection intervals in Chapter 5.

A. Remove Accelerometer (See figure 201)
 (1) Remove nose compartment doors.
 (2) Remove electrical power from aircraft.
 (3) Locate accelerometer on left upper quarter of frame 2. Remove electrical connector. Note facing direction.
 (4) Remove attaching parts and accelerometer from aircraft.

B. Install Accelerometer (See figure 201)
 (1) Assure that accelerometer is correct part number.
 (2) Position accelerometer with placarded arrow up or, on certain aircraft, with the word TOP on top.
 (3) Secure accelerometer to frame 2 with attaching parts.
 (4) Connect electrical connector to accelerometer.
 (5) Install nose compartment doors.
 (6) Restore electrical power to aircraft.

2. Adjustment/Test

A. Functionally Test Accelerometer (Aircraft Equipped with P/N 2380071 Accelerometer)
 (1) Remove accelerometer from aircraft.
 (2) Place accelerometer on bench in its normal mounting position.
 (3) Check resistance between pins A & B and E & F of connector.
 (4) Resistance shall be less than 20 ohms in either check.
 (5) If resistance is in excess of 20 ohms, remove cover from accelerometer.
 (6) Add washers (AN960-3L) as required between stop plate and housing base. This allows accelerometer mass to be positioned downward in order to obtain a resistance of less than 20 ohms.
 (7) Recheck resistance between pins A & B and E & F; resistance shall be less than 20 ohms.
 (8) Install cover on accelerometer.
 (9) Connect ohmmeter to pins A & B and, with accelerometer in normal mounting position, rotate 40° (±10) CW then CCW.
 (10) The resistance reading shall not increase. Beyond 40° (±10°) CW or CCW, the resistance reading shall increase smoothly.
 (11) Pitch the accelerometer 40° (±10°) forward and aft.
 (12) The resistance reading shall not increase. Beyond 40° (±10°) forward and aft the resistance reading shall increase smoothly.
 (13) Repeat steps (9) through (12) with ohmmeter connected to pins E & F.

B. Functionally Test Accelerometer (Aircraft Equipped with P/N 6608283-2 Accelerometer)

(1) Remove accelerometer from aircraft.
(2) Place accelerometer on bench in its normal mounting position.
(3) Using a volt-ohmmeter, check resistance between pins A & E of connector.
(4) Resistance shall be less than 0.4 ohm.
(5) If the resistance is more than 0.4 ohm, ensure that the accelerometer is on a flat, level surface and is in the vertical plane.
(6) Rotate accelerometer forward until volt-ohmmeter reads infinity. Measure angle from vertical to point at which volt-ohmmeter reads infinity (point of reference on accelerometer is mounting plate). The angle shall be between 53° and 63.5°.

NOTE: If the angle at which the volt-ohmmeter indicates infinity is not within tolerance, repeat the test and ensure that the accelerometer is in the horizontal plane.

(7) Repeat step (6), rotating the accelerometer aft. Return accelerometer to its normal mounting position.
(8) Rotate accelerometer to right until volt-ohmmeter reads infinity. Measure angle from horizontal to point at which volt-ohmmeter reads infinity (point of reference on accelerometer is bottom edge of mounting plate). The angle shall be between 53° and 63.5°.

NOTE: If the angle at which the volt-ohmmeter indicates infinity is not within tolerance, repeat test and ensure that the accelerometer is in the vertical plane.

(9) Repeat step (8), rotating accelerometer to left.
(10) If accelerometer does not function as described, it must be replaced with a new accelerometer.

C. Functionally Test Accelerometer (Aircraft Equipped with P/N 6608283-1 Accelerometer)

(1) Remove accelerometer from aircraft.
(2) Place accelerometer on bench in its normal mounting position.
(3) Using a volt-ohmmeter, check resistance between pins B & C and E & F of connector.
(4) Resistance should be less than 0.2 ohm.
(5) If resistance is more than 0.2 ohm, ensure that accelerometer is on a flat, level surface and is in the vertical plane.
(6) Measure resistance between B & C. Rotate accelerometer forward until volt-ohmmeter reads infinity. Measure angle from vertical to point at which volt-ohmmeter reads infinity (point of reference on accelerometer is mounting plate). The angle shall be between 53° and 63.5°.

NOTE: If the angle at which the volt-ohmmeter indicates infinity is not within the tolerance, repeat test and ensure that accelerometer is in the horizontal plane.

(7) Repeat step (6) while rotating accelerometer aft from vertical. Return accelerometer to its normal mounting position.

(8) Rotate accelerometer to right until volt-ohmmeter reads infinity. Measure angle from horizontal to point at which volt-ohmmeter reads infinity (point of reference on accelerometer is bottom of mounting plate). The angle shall be between 53° and 63.5°.

NOTE: If the angle at which the volt-ohmmeter indicates infinity is not within tolerance, repeat test and ensure that accelerometer is in the vertical plane.

(9) Return accelerometer to its normal mounting position and repeat step (8) rotating accelerometer to left.

(10) Repeat steps (6) through (9) while probing pins E & F of connector.

(11) If accelerometer does not function as described, it must be replaced by a new accelerometer.

3. Inspection/Check

A. Operational Check of Accelerometer

(1) Remove LH nose compartment access door.

(2) Remove accelerometer attaching parts and have one person hold accelerometer in its normal position.

(3) Connect external power to aircraft and set Battery Switch(es) ON. Assure that 28 ±0.5 vdc is available.

CAUTION: LIMIT THIS OPERATIONAL CHECK TO A SHORT DURATION SO AS NOT TO OVERHEAT THE COMPONENTS.

(4) Set LH Stall Warning Switch to ON.

(5) Adjust horizontal stabilizer position until a neutral feel of control column is obtained.

(6) Attach a spring scale (0 to 100 pounds) to control column.

(7) Holding accelerometer in its normal position, move LH stall vane up slowly until pusher is actuated. Check force at control column is 40-80 pounds.

(8) Rotate accelerometer 180° (upside down from normal position).

(9) Check that force at control column is reduced to 20 pounds or less.

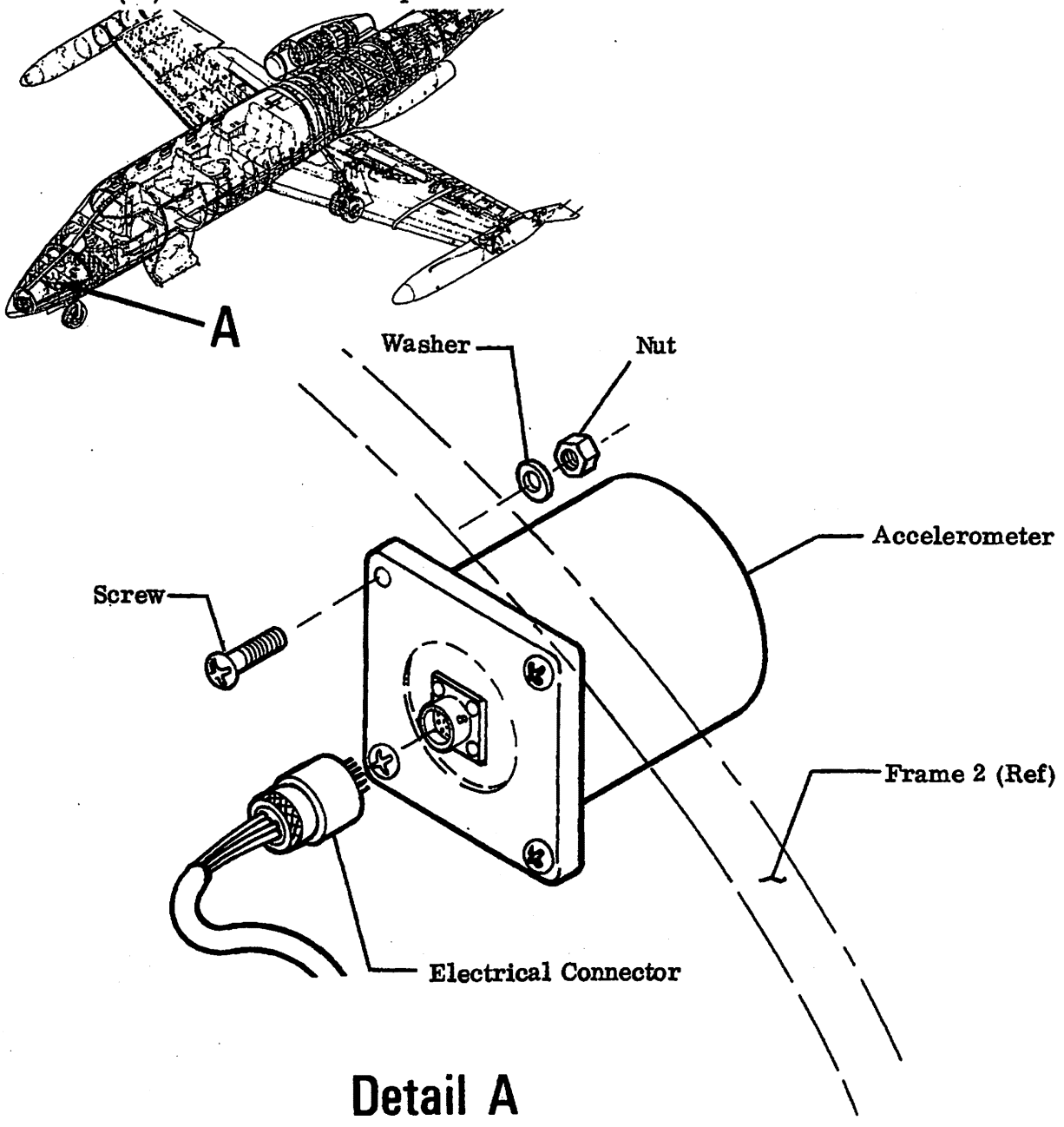
(10) Return accelerometer to normal position and set LH Stall Warning Switch to OFF.

(11) Repeat steps (1) through (10) using the RH Stall Warning Switch and RH Stall Vane.

(12) If the above conditions are not met, perform functional test of accelerometer and adjust per functional test or replace as required.

NOTE: Resistance of accelerometer P/N 2380071 is adjustable.

(13) Set Battery Switch(es) to OFF and disconnect external power.
(14) Install nose compartment access doors.



Detail A

Accelerometer Installation
Figure 201

9-201B

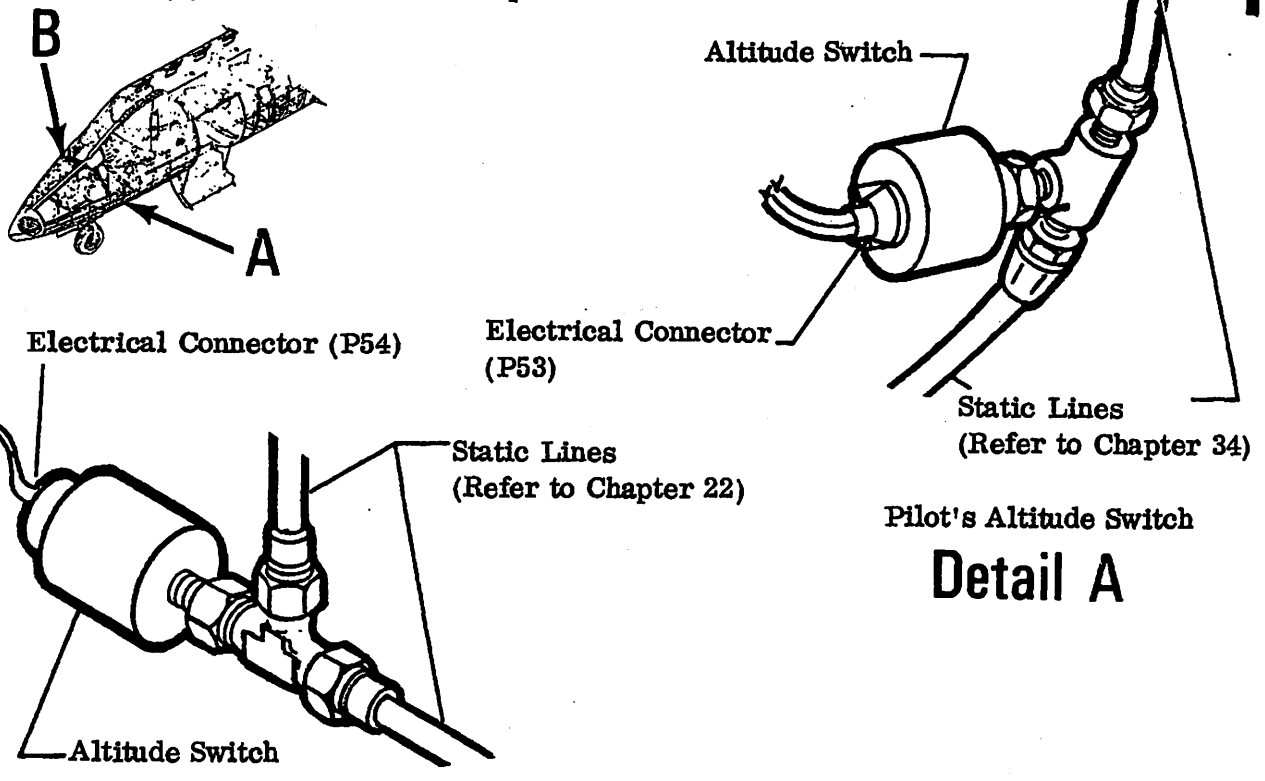
STALL WARNING ALTITUDE SWITCH - MAINTENANCE PRACTICES

1. Removal/Installation

NOTE: The altitude pressure switches must be functionally checked in accordance with current inspection intervals in Chapter 5.

A. Remove Altitude Pressure Switch (See figure 201)
 (1) Remove electrical power from aircraft.
 (2) Remove nose compartment access doors.
 (3) Gain access to applicable altitude switch installation.
 (4) Disconnect electrical connector from altitude switch.
 (5) Loosen and remove altitude switch from static lines.

B. Install Altitude Pressure Switch (See figure 201)
 (1) Install and secure altitude pressure switch to static line.
 (2) Connect electrical connector to altitude switch.
 (3) Install nose compartment access doors.
 (4) Restore electrical power to aircraft.



Stall Warning Altitude Switch Installation
Figure 201

14-28C-10

2. Inspection/Check

A. Altitude Switch Operational Check

(1) Set both Stall Warning Switch(es) on.

(2) Lower both stall warning vanes so that angle-of-attack indicator is midway through the green band.

(3) Position flaps to full up.

CAUTION: TO AVOID DAMAGE TO THE AIR DATA SENSOR, ALWAYS APPLY EQUAL VACUUM TO THE RH PITOT HEAD, SHOULDER STATIC PORTS, AND COPILOT'S STATIC SYSTEM.

(4) Connect pitot static test set (1811F - Barfield Instrument Corp. or equivalent) to shoulder static ports and slowly apply vacuum.

(5) Observe pilot's angle-of-attack indicator and the tester altimeter. The pilot's angle-of-attack indicator shall abruptly move toward the yellow area (stall) when tester altimeter reads 22,500 (±750) feet. Copilot's indicator shall not move.

(6) Slowly bleed off vacuum until field elevation is reached. Remove tester from shoulder ports.

(7) Connect tester to RH forward and LH aft static ports and slowly apply vacuum.

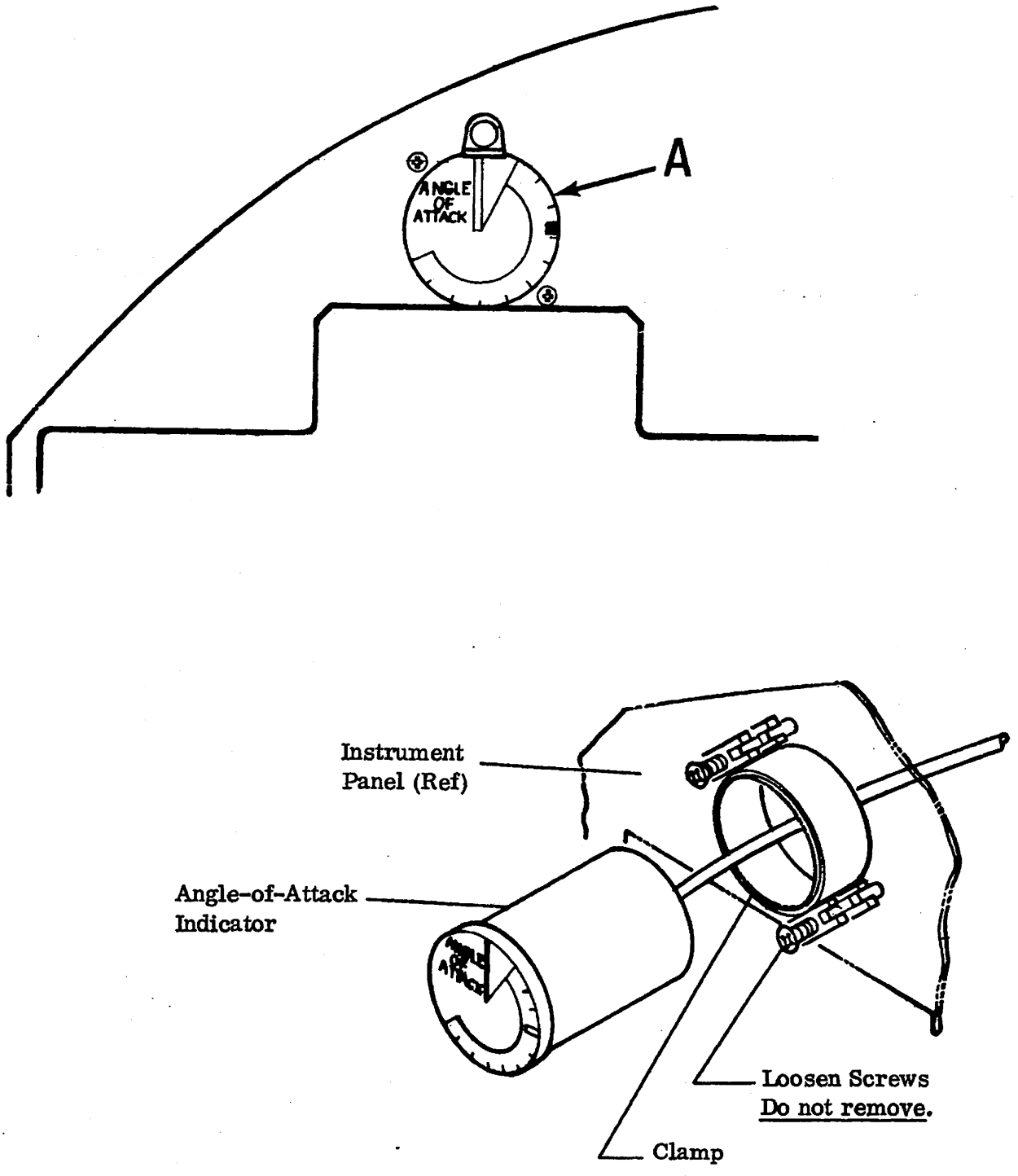
(8) Observe copilot's angle-of-attack indicator and the tester altimeter. The copilot's angle-of-attack indicator shall abruptly move toward the yellow area (stall) when the tester altimeter reads 22,500 (±750) feet. Pilot's angle-of-attack indicator shall not move.

(9) Slowly bleed off vacuum until field elevation is reached. Remove tester from aircraft.

ANGLE-OF-ATTACK INDICATOR - MAINTENANCE PRACTICES

1. Removal/Installation
 - A. Remove Angle-of-Attack Indicator (See figure 201)
 - (1) Remove electrical power from aircraft.
 - (2) Loosen mounting screws on either side of indicator face.
 - (3) Withdraw indicator.
 - (4) Disconnect electrical connector from back of indicator.
 - (5) Remove indicator from panel.
 - B. Install Angle-of-Attack Indicator (See figure 201)
 - (1) Connect electrical connector to indicator.
 - (2) Install indicator in panel and secure with attaching parts.
 - (3) Restore electrical power to aircraft.

2. Inspection/Check
 - A. Angle-of-Attack Indicator Functional Test (See figure 201)
 - (1) Perform Stall Warning Functional Test per applicable instructions. (Refer to 27-31-00.)



Detail A

Angle-of-Attack Indicator Installation
Figure 201

16-37A

STALL WARNING FLAP BIAS SWITCH - MAINTENANCE PRACTICES

1. Removal/Installation
 A. Remove Stall Warning Flap Bias Switch (See figure 201)
 (1) Position flaps down.
 (2) Remove electrical power from aircraft.
 (3) Locate switch assembly in the flap well at WS 74.
 (4) Remove flap sector access panel just forward of switch assembly installation.
 (5) Remove switch actuating bellcrank and mounting plate attaching parts.
 (6) Disconnect electrical wiring from wiring harness.
 (7) Remove flap switch assembly from aircraft.
 B. Install Stall Warning Flap Bias Switch (See figure 201)

 NOTE:
 - Refer to applicable wiring diagram.
 - Flap bias switch and bellcrank assembly is a factory pre-set component. Do not disturb bellcrank/shaft relationship or switch adjusting screws.

 (1) Connect electrical wiring to wiring harness.
 (2) Install switch mounting plate attaching parts.
 (3) Install bellcrank attaching parts.
 (4) Assure that ample clearance exists between wiring and moving parts.
 (5) Perform applicable Stall Warning Functional Test. (Refer to 27-31-00.)

2. Inspection/Check (See figure 202)
 A. Check Continuity of LH Stall Warning Flap Bias Switch
 (1) Disconnect electrical power from aircraft.
 (2) Locate and disconnect electrical connector P421 on the aft side of the LH wheel well at FS 364.
 (3) Fabricate and install a patch cable between P421 electrical connector halves with labeled wire ends from Stall Warning Flap Bias Switch pins 15, 53, and 52. Each wire end shall be fitted with an ohmmeter probe jack.
 (4) Apply 28 vdc to aircraft.
 (5) Position flap to full up.
 (6) Remove 28 vdc power from aircraft.
 (7) Connect one ohmmeter probe to a clean aircraft ground and the other probe to P421 pins 15, 53, and 52 in sequence. The ohmmeter shall indicate an open Stall Warning Flap Bias Switch circuit at each test point.
 (8) Apply 28 vdc to aircraft and lower flaps to 8°.
 (9) Remove 28 vdc from aircraft.

(10) Connect one ohmmeter probe to a clean aircraft ground and the other probe to P421 pins 15, 53, and 52 in sequence. The ohmmeter shall indicate Stall Warning Flap Bias Switch at pin 52. Pins 15 and 53 shall be open circuits.

(11) Apply 28 vdc power to aircraft and lower flaps to 20°.

(12) Remove 28 vdc power from aircraft.

(13) Connect one ohmmeter probe to a clean aircraft ground and the other probe to P421 pins 15, 53, and 52 in sequence. The ohmmeter shall indicate Stall Warning Flap Bias Switch at pins 15 and 53. Pin 52 shall be an open circuit.

(14) Apply 28 vdc power to aircraft and lower flaps to full down.

(15) Remove 28 vdc power from aircraft.

(16) Connect one ohmmeter probe to a clean aircraft ground and the other probe to P421 pins 15, 53, and 52 in sequence. The ohmmeter shall indicate Stall Warning Bias Switch continuity at pins 15, 53, and 52.

(17) Connect electrical connectors and secure as required.

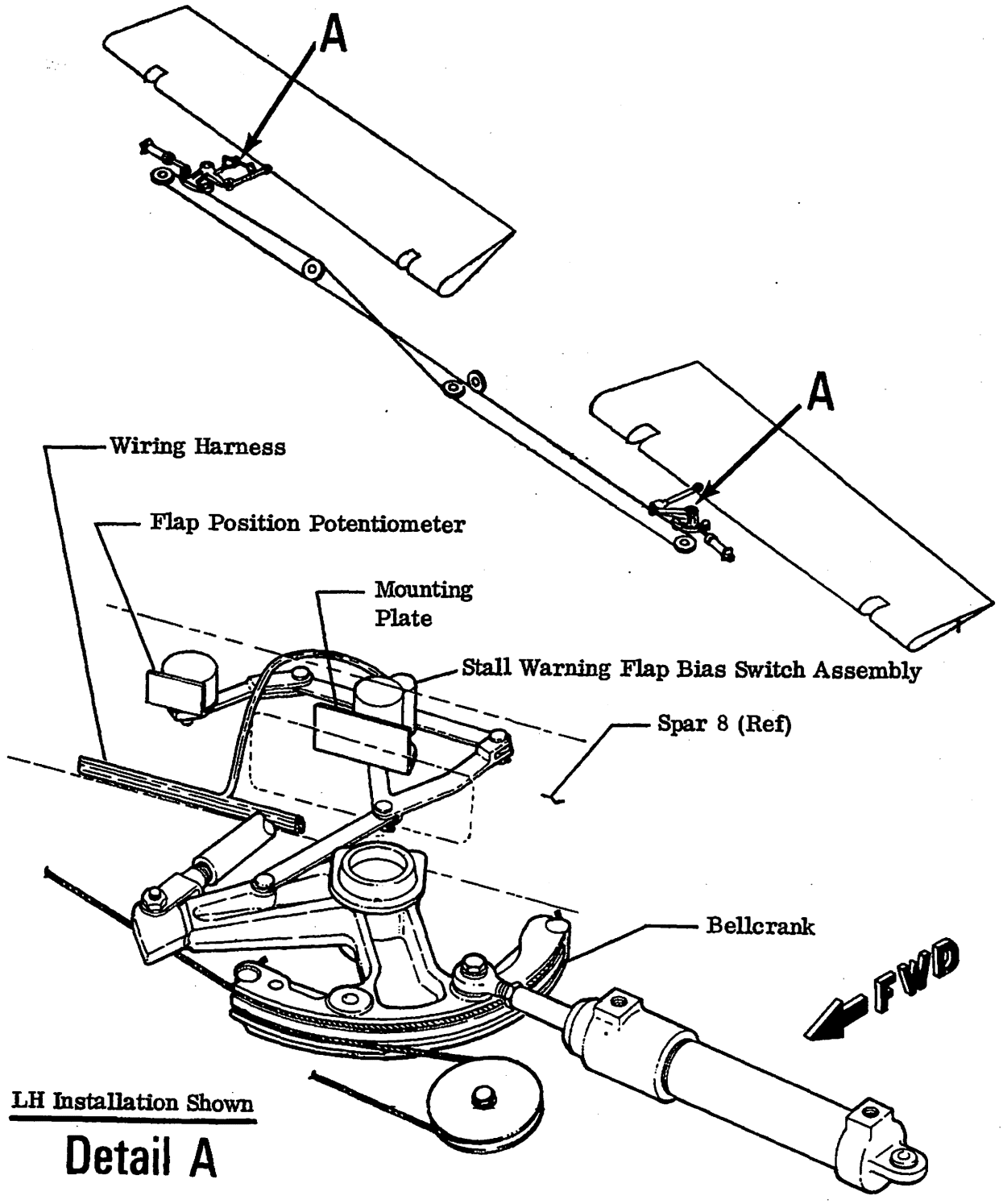
B. Check Continuity of RH Stall Warning Flap Bias Switch

(1) Disconnect electrical power from aircraft.

(2) Locate and disconnect electrical connector P422 on the aft side of the RH wheel well at FS 364.

(3) Fabricate and install a patch cable between P421 electrical connector halves with labeled wire ends from Stall Warning Flap Bias Switch, pins 5, 6, and 15. Each wire end shall be fitted with an ohmmeter probe jack.

(4) Perform test steps 2.A.(4) through (17) listed under LH Stall Warning Flap Bias Switch using pins, 5, 6, and 15.



Stall Warning Flap Bias Switch Installation
Figure 201

	SWITCH STAGES		
Flap Position	Switch 1	Switch 2	Switch 3
Full UP (0°)	-	-	-
Between 0° and 13°	X	-	-
Between 13° and 25°	X	X	-
Between 25° and 40° (DOWN)	X	X	X

X denotes closure to ground.

Stall Warning Flap Bias Switch Operation
Figure 202

STALL WARNING VANE PLATE - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLATION

A. Remove Stall Warning Vane Plate (See figure 201.)

(1) Remove screws from vane plate.

(2) Pry vane plate from aircraft using care not to damage aircraft skin.

B. Install Stall Warning Vane Plate (See figure 201.)

(1) Mask off vane plate mating surface.

(2) Thoroughly clean surfaces to be cemented using methyl ethyl ketone (MEK). Wipe surface dry with a clean cotton cloth.

(3) Mix 100 parts by weight of EC-2216B (white) to 140 parts by weight of EC-2216A (gray). Thoroughly mix components until completely blended into a uniform gray paste. This adhesive system is manufactured by the 3M Company.

(4) Apply adhesive 0.002 inch minimum thickness over both mating surfaces using a spatula. Place new vane plate, P/N 2514313-1 (LH), 2514313-2 (RH) over adhesive surface and secure with existing screws. Allow adhesive to cure 24 hours at room temperature. An accelerated full cure can be attained by heating for two hours at 180°F or one hour at 250°F.

2. ADJUSTMENT/TEST

A. Mark New Vane Plates (Aircraft 25-061 and 25-070 thru 25-289 not equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers.")

NOTE: If stall warning pusher and shaker actuation is not within tolerance specified in 27-31-00 Maintenance Practices, paragraph 1.A., 1.D., 1.G., or 1.I., replace vane plate per paragraph 1. above.

(1) Position a 2-inch wide strip of masking tape over new vane plate and mark an arc on the tape at the trailing edge of the vane using a ball-point pen.

(2) Perform "Functional Test of Stall Warning System" per 27-31-00 Maintenance Practices, paragraph 1.A., 1.D., 1.G., or 1.I., adding marks to tape per figure 201 at actuation points.

(3) Repeat functional test to check marks on tape.

(4) Scribe marks except arc through masking tape.

B. Check Vane Plate Markings (Aircraft 25-290 and Subsequent and prior Aircraft equipped with AAK 79-10 or AMK 83-5, "Installation of Wing Fences, Stall Strips, and Boundary Layer Energizers.")

NOTE: ° If stall warning pusher and shaker actuation is not within the tolerance specified in 27-31-00 Maintenance Practices, paragraph 1.G. or 1.I., the vane plate marks can be verified for accuracy using the following procedure. Installation and marking of a new vane plate may be necessary.

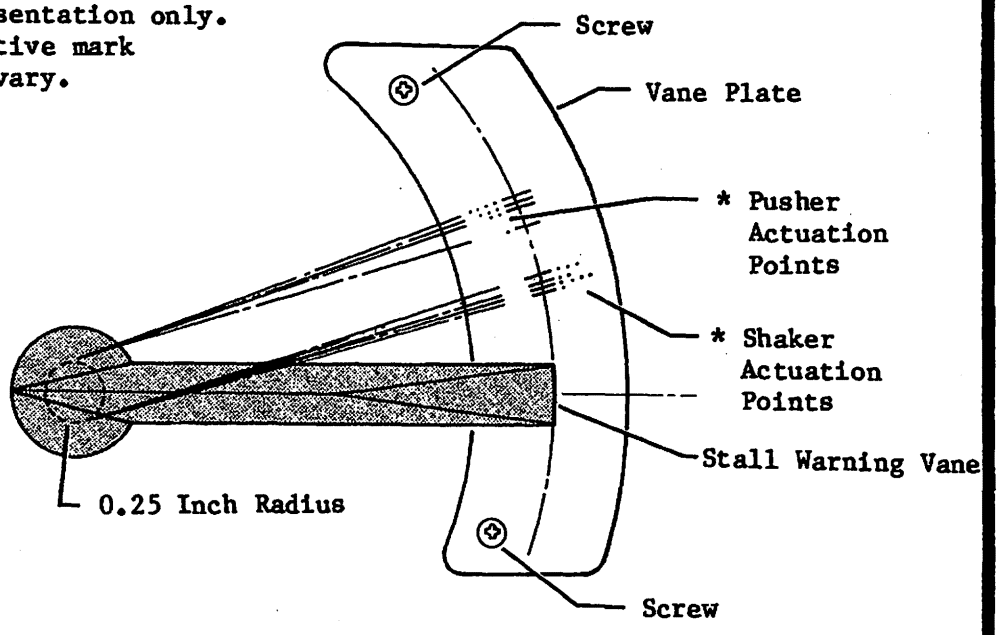
NOTE: ° Mylar templates, P/N 5471102-5 (LH) and P/N 5471102-6 (RH), are provided to verify mark location and to assist in remarking the plate while the plate is installed on the airplane. When correctly marked, the vane plate provides a reference for easily testing the stall warning system for calibration acuracy on subsequent maintenance inspections.

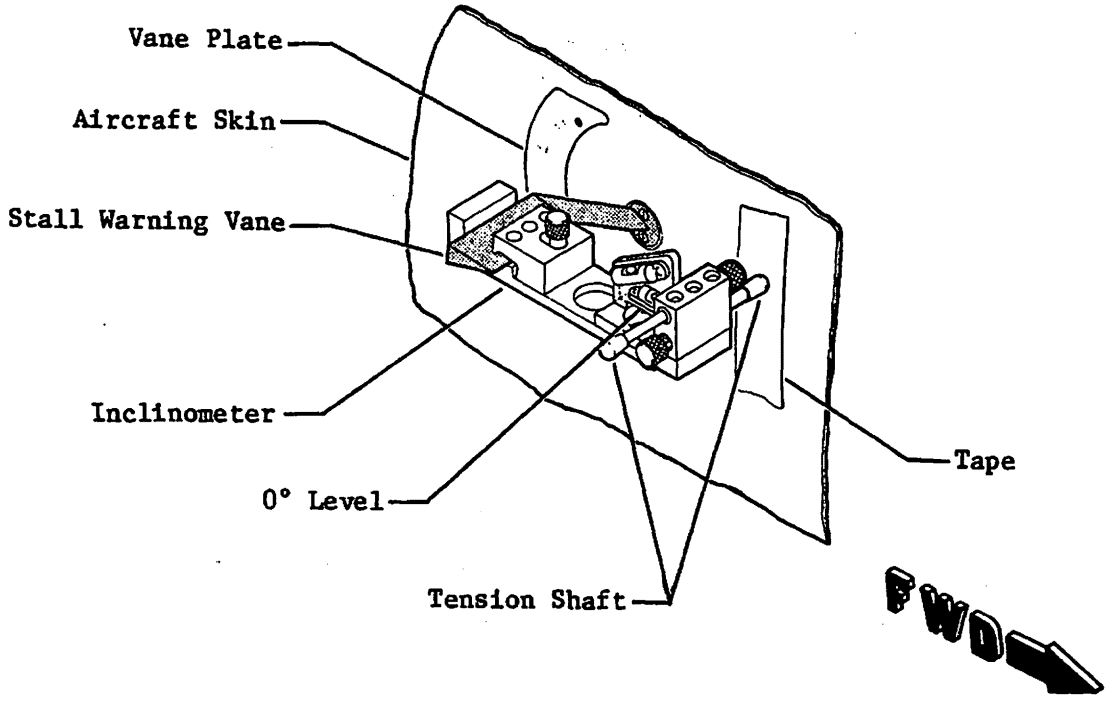
(1) Using jacks, level the airplane. (Refer to Chapter 8.)
(2) If vane plate is not marked by an arc from top to bottom under trailing edge of vane, use a scribe taped to training edge of vane to carefully scribe an arc for reference.
(3) Place a piece of tape forward of stall vane so that rubber-tipped tension shaft of inclinometer will not mar painted surface. Install vane positioning inclinometer, P/N T3500002-600, on vane and position vane to zero degrees. Secure tool finger-tight only.
(4) Place vane plate marking template over vane plate and align it with arc on the vane plate. Next, position mylar template along arc until template reference line is aligned with top edge of vane. Secure mylar template in this position.
(5) Remove vane position inclinometer for better access to mylar.
(6) Compare existing marks on vane plate with the corresponding marks on the template.
(7) If mark positions are not in agreement, remove old vane plate and install a new, blank vane plate. (Refer to paragraph (1) above.)
(8) Position the mylar template as described in steps (1) through (4) above.
(9) Before final scribing of marks, verify that shaker and pusher actuation occurs within prescribed tolerance of template marks.

NOTE: If all test points are uniformly in error, it is likely that the template is mislocated. If the error varies, it is likely that the computer is misadjusted. If computer misadjustment is suspected, perform the ground calibration procedure per 27-31-00 Maintenance Practices, paragraph 1.M. prior to marking the vane.

(10) When correct template location and computer calibration are assured, carefully scribe reference marks through template onto plate.
(11) Remove mylar tempate and scribe appropriate lines and symbols through reference marks on step (9).
(12) Perform the Functional Test per 27-31-00, paragraph 1.G., or 1.I., steps (21) through (33); assure that actuation of the shaker and pusher occurs at the correct position.

* This is a representation only. Actual and relative mark locations will vary.





15-14A
14-166A

Stall Warning Vane Plate Installation and Marking
Figure 201

ANGLE-OF-ATTACK VANE - MAINTENANCE PRACTICES

1. REMOVAL/INSTALLTION

WARNING: AVOID CONTACT WITH HOT ANGLE-OF-ATTACK VANES.

NOTE: ° Removal and installation procedures for both angle-of-atack vanes are identical.

° Replacement of the stall transducer vane is permitted in the event of a failed heater. Physical damage to the vane requires total transducer replacement because of possible damage to the transducer shaft, bearing, and potentiometer. Vane replacement requires functional test to verify that shaker and pusher actuation occurs within the prescribed tolerance of the vane plate marks.

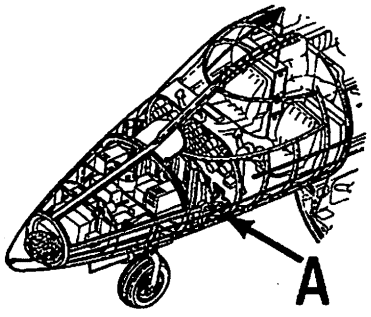
A. **Remove Angle-of-Attack Vane** (See figure 201.)
 (1) Remove electrical power from aircraft.
 (2) Ensure that angle-of-attack vanes are not hot.
 (3) Remove attaching parts securing angle-of-attack vane to angle-of-attack transducer assembly.

CAUTION: ° DO NOT PRY ANGLE-OF-ATTACK VANE OFF, OR ALLOW IT TO COCK TO ONE SIDE WHILE REMOVING AS DAMAGE TO THE TRANSDUCER ASSEMBLY COULD RESULT.

° A SLIGHT FORWARD AND AFT MOTION WHILE APPLYING PRESSURE TO VANE WILL DISENGAGE VANE FROM TRANSDUCER ASSEMBLY.

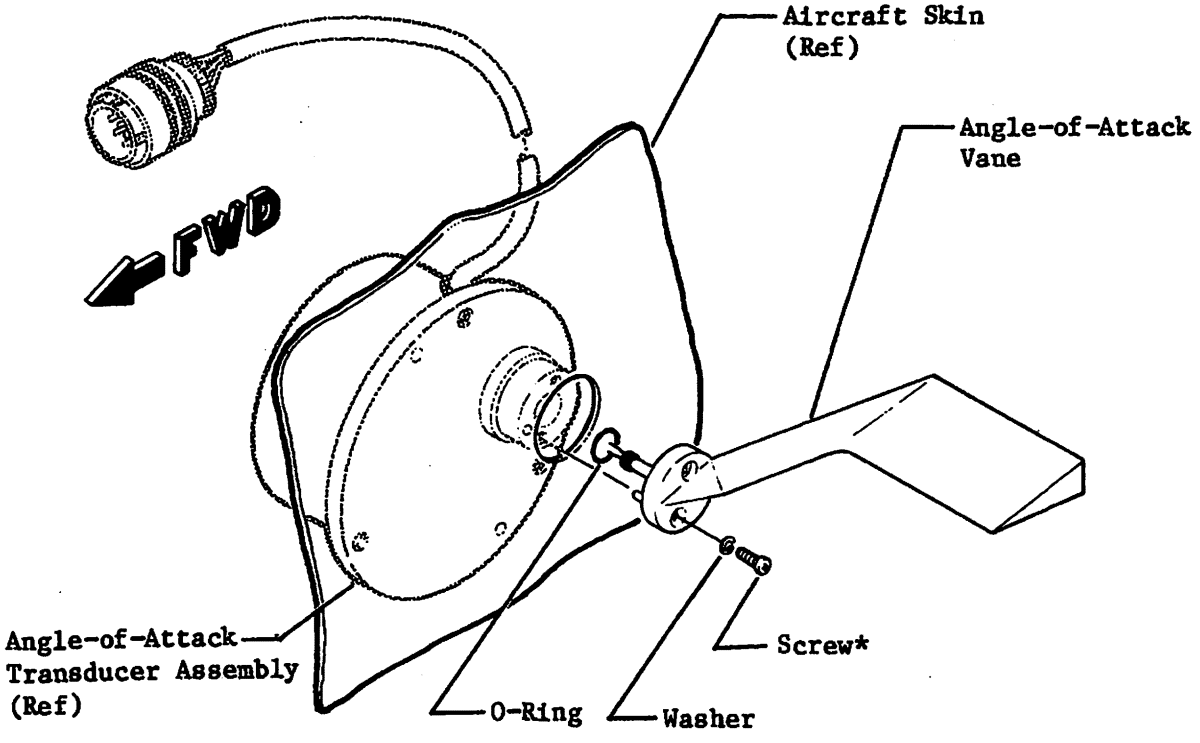
 (4) Carefully remove vane from transducer assembly.

B. **Install Angle-of-Attack Vane** (See figure 201.)
 (1) Position angle-of-attack vane at its appropriate location on angle-of-attack transducer assembly, and secure with attaching parts. Alternately tighten screws until vane is seated firmly in place.
 (2) Restore electrical power to aircraft.
 (3) Perform stall warning system functional test. (Refer to 27-31-00 and perform applicable functional test.)
 (4) Restore aircraft to normal.



CAUTION: ° DO NOT PRY ANGLE-OF-ATTCK VANE OFF, OR ALLOW IT TO COCK TO ONE SIDE WHILE REMOVING AS DAMAGE TO THE TRANSDUCER ASSEMBLY COULD RESULT.

° A SLIGHT FORWARD AND AFT MOTION WHILE APPLYING PRESSURE TO VANE WILL DISENGAGE VANE FROM TRANSDUCER ASSEMBLY.



*Torque 10-14 inch-pounds

Detail A

Angle-of-Attack Vane Installation
Figure 201

HORIZONTAL STABILIZER CONTROL SYSTEM - DESCRIPTION AND OPERATION

1. Description (See figure 1)
 A. The horizontal stabilizer control system provides pitch trim control in normal and emergency modes.
 B. On Aircraft 25-061 and 25-070 thru 25-341, except 25-337, NOT modified for Horizontal Stabilizer Trim and Autopilot Improvement, the horizontal stabilizer control system consists of the horizontal stabilizer, an actuator, a stabilizer trim position indicator, Pitch Trim and Emergency Pitch Trim Switches, and the Pilot's and Copilot's Trim and Trim Arming Switches, and utilizes relays in the switch panel.
 C. On Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement, the horizontal stabilizer control system consists of the horizontal stabilizer, a two-speed actuator, a stabilizer trim position indicator, a pitch trim speed controller, Pitch Trim and Emergency Pitch Trim Switches, a Pitch Trim Monitor Test Switch, the Pilot's and Copilot's Trim and Trim Arming Switches, a pitch trim disable terminal board, an overload sensor and relay assembly, TAKE-OFF TRIM and PITCH TRIM OVSP warning lights, a squat switch, and flap switches and utilizes relays in the switch panel.

2. Component Description
 A. The movable horizontal stabilizer serves as the pitch trim controller. The stabilizer is attached to the vertical stabilizer at two points. The aft attachment is the pivot point and consists of a large hinge pin inserted through heavy bearings in the horizontal and vertical stabilizers. The forward attach point is connected to a heavy-duty electrical actuator. The bottom end of the actuator is secured to the vertical stabilizer forward spar.
 B. The screwjack-type actuator adjusts the horizontal stabilizer to provide aircraft longitudinal trim and is installed on vertical stabilizer spar No. 2 at WL 96.
 C. The pitch trim indicator is located on the pedestal.
 D. The pitch trim speed controller is located in the vertical stabilizer forward of spar 2 at WL 85.
 E. The Pitch Trim Switch, located on the switch panel, is a three-position toggle switch used to select normal or emergency modes of operation or to turn the pitch trim system off and is marked P TRIM (primary trim) NORM, OFF, and EMER.
 F. The Emergency Pitch Trim Switch, located on the trim indicator panel on the pedestal, is a three-position toggle switch used to manually change pitch when the Pitch Trim Switch is set to EMERGENCY and is marked EMERGENCY, NOSE UP, and NOSE DOWN.
 G. The Pitch Trim Monitor Test Switch, located on the pedestal, is a two-position, momentary-contact toggle switch.

H. The Pilot and Copilot's Trim and Trim Arming Switches are mounted on the control wheels and are marked NOSE DOWN, NOSE UP, LWD (left wing down), and RWD (right wing down).

I. The pitch trim disable terminal board is located on the bottom forward end of the radar support assembly, forward of the instrument panel.

J. The overload sensor and relay assembly is located in the tailcone on frame 25. (Refer to 27-40-06).

K. The TAKE-OFF TRIM and PITCH TRIM OVSP warning lights are located on the LH side of the glareshield. The TAKE-OFF TRIM warning light provides visual warning of horizontal stabilizer pitch angles which are inappropriate for take-off. The PITCH TRIM OVSP warning light provides visual warning of horizontal stabilizer actuator travel at high speed when flaps are between 3° and full up.

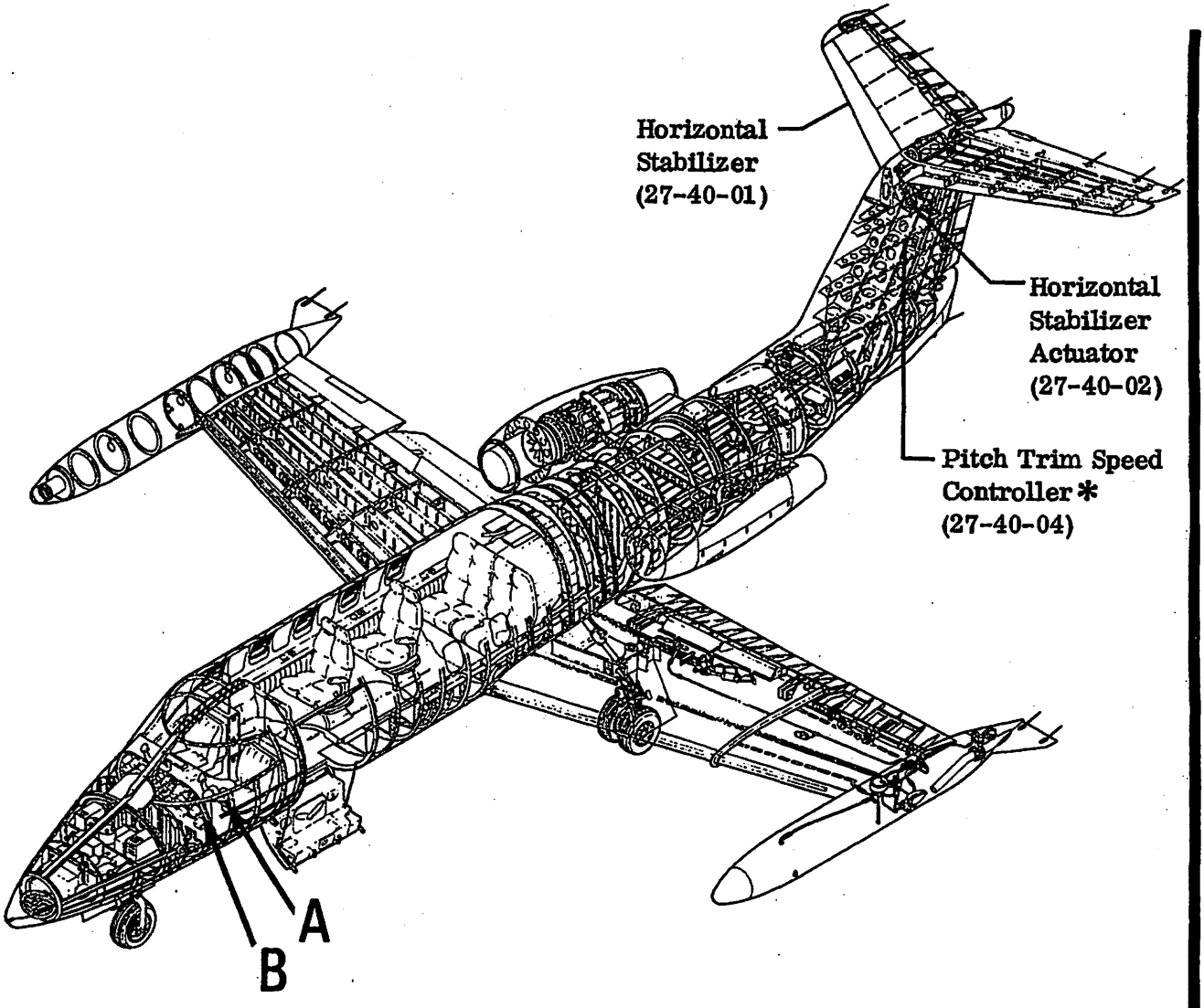
3. Operation (See figure 2)

NOTE: Normal and emergency horizontal stabilizer operation is described below. A third mode of operation (autopilot) is described in Chapter 22.

A. Aircraft 25-061 and 25-070 thru 25-341, except 25-337, NOT modified for Horizontal Stabilizer Trim and Autopilot Improvement.

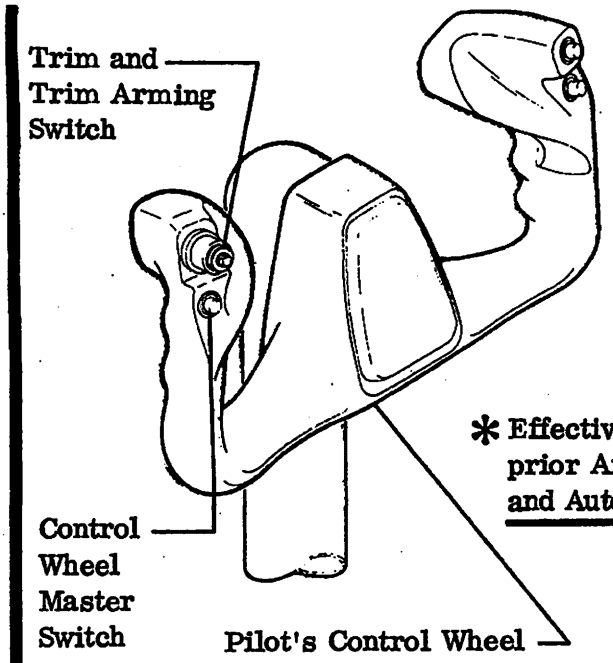
(1) With the Pitch Trim Switch set to normal (NORM), a standby ground circuit is completed through the Pitch Trim Switch to the de-energized contacts of the trim control relay. Twenty-eight vdc is applied from the PITCH circuit breaker to the Pilot's Trim and Trim Arming Switch. When the Pilot's Trim and Trim Arming Switch is depressed and set to either NOSE UP or NOSE DN, 28 vdc is applied to the pitch trim actuator primary motor, the pilot's authority relay, and the trim control relay. The pilot's authority relay energizes to remove power from the Copilot's Trim and Trim Arming Switch while the trim control relay energizes to complete the ground circuit to the pitch trim actuator primary motor. Operation of the copilot's trim system is identical to the pilot's trim system, except 28 vdc is applied through the de-energized contacts of the pilot's authority relay to the Copilot's Trim and Trim Arming Switch. During normal trimming operation, the autopilot will be disengaged.

(2) With the Pitch Trim Switch set to EMER, the ground source for the normal pitch trim motor is removed. This makes the pitch trim function of the Pilot's and Copilot's Trim and Trim Arming Switches inoperative. The autopilot will disengage when 28 vdc (either nose up or nose down) is applied through the Emergency Pitch Trim Switch to the pitch trim actuator secondary motor. During emergency operation, a ground circuit is completed through the contacts of the Pitch Trim Switch to the pitch trim actuator secondary motor. Twenty-eight vdc

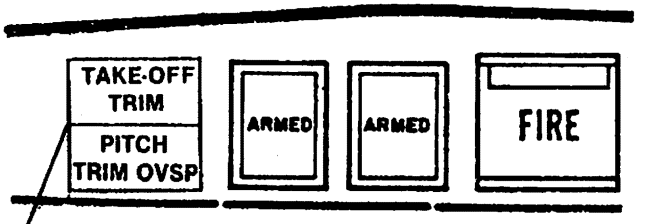


* Effective on 25-337, 25-342 and Subsequent and prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement.

Horizontal Stabilizer Control System Component Locator
Figure 1 (Sheet 1 of 2)

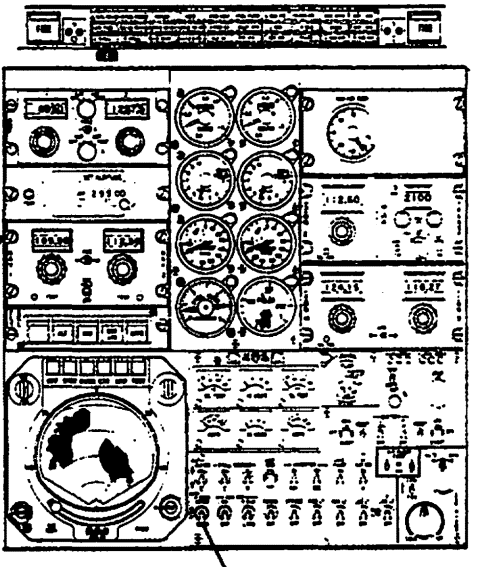


Detail A



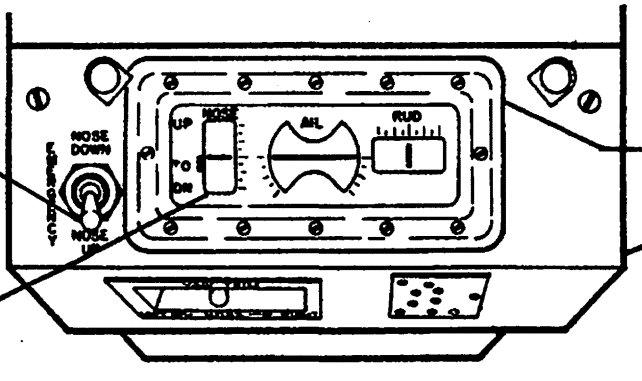
Take-Off Trim and
Pitch Trim Overspeed Warning Lights
(Amber)*

* Effective on Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement.



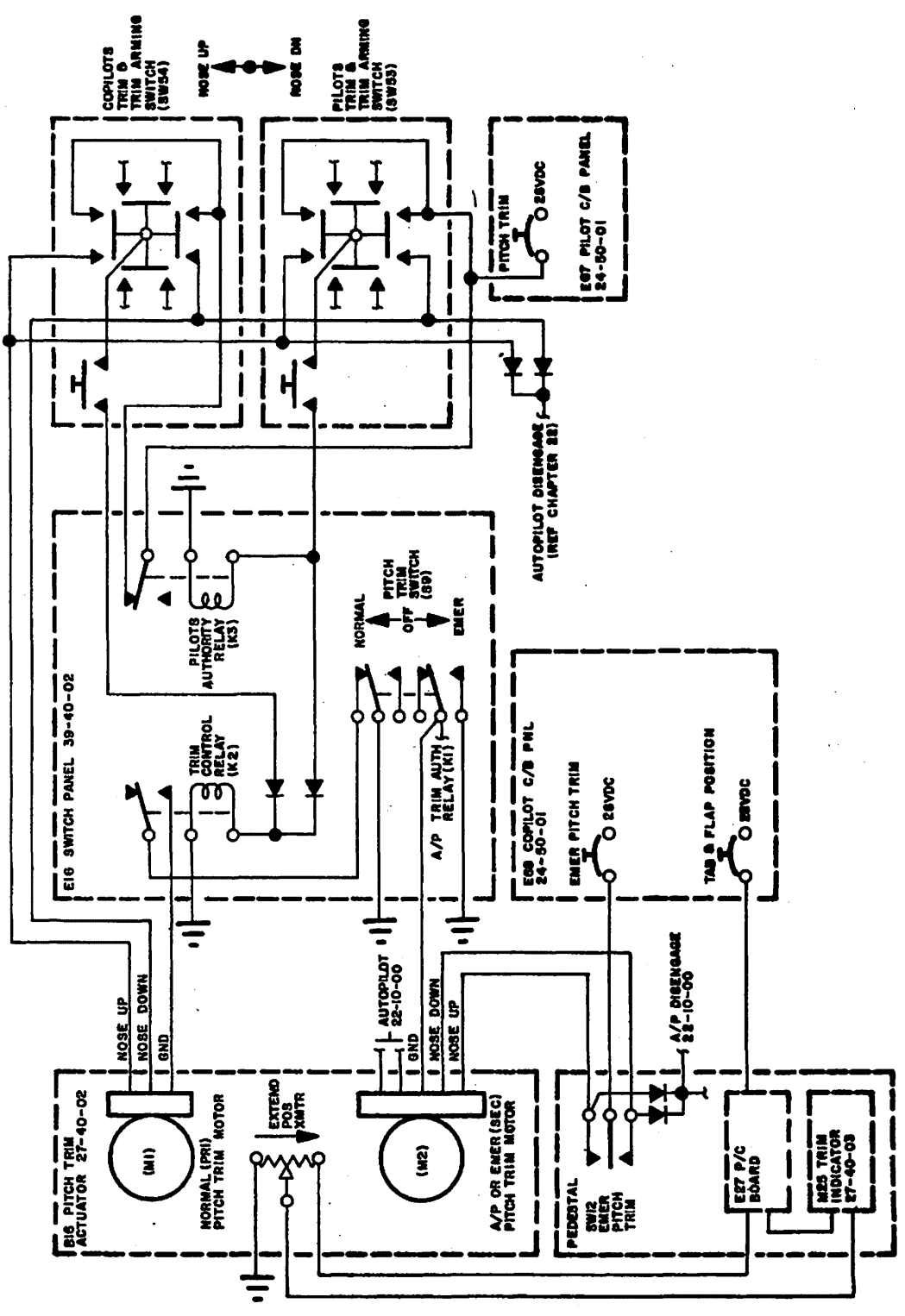
Detail B

Pitch Trim
Switch



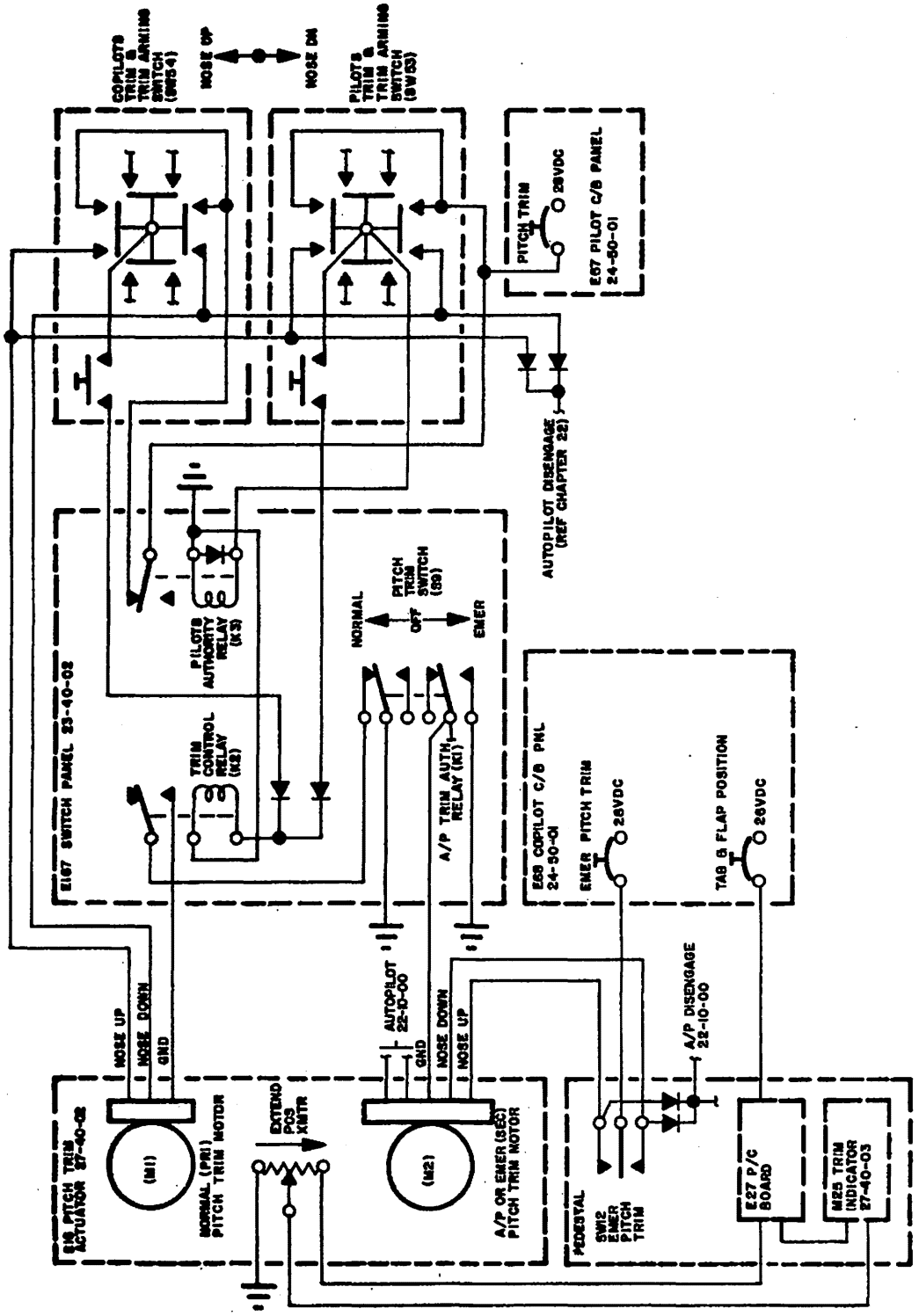
Horizontal Stabilizer Control System Component Locator
Figure 1 (Sheet 2 of 2)

A8-51B



Pitch Trim Electrical Control Schematic
Figure 2 (Sheet 1 of 3)

EFFECTIVITY: Aircraft 25-061, 25-070 thru 25-201 NOT modified for Horizontal Stabilizer Trim and Autopilot Improvement



Pitch Trim Electrical Control Schematic
Figure 2 (Sheet 2 of 3)

EFFECTIVITY: Aircraft 25-202 thru 25-341, except 25-337, NOT modified for Horizontal Stabilizer Trim and Autopilot Improvement

relay will not be present due to the open PITCH circuit breaker. After the malfunction has been corrected and the PITCH circuit breaker reset, the relay will energize.

(2) Emergency Operation

(a) With the Pitch Trim Switch set to EMER, the Pilot's and Copilot's Trim and Trim Arming Switches are rendered inoperative. Stabilizer trim is accomplished by setting the Emergency Pitch Trim Switch to either NOSE UP or NOSE DOWN. This directs 28 vdc to the pitch trim actuator secondary motor which operates at a fixed low speed. During emergency operation, a ground circuit is completed through the contacts of the Pitch Trim Switch to the pitch trim actuator secondary motor.

(b) The horizontal stabilizer control circuitry is biased through the right flap switch which closes as the flaps are extended through 3°, shifting the stabilizer actuator operational mode from low to high speed. Actuator speed monitoring is accomplished through two magnetic pickups in the primary motor.

(3) Normal and Emergency Operation

(a) Operation of the horizontal stabilizer control system in either normal or emergency mode will disengage the autopilot.

(b) Depressing the Control Wheel Master Switch energizes two trim disable relays in the pitch trim actuator, rendering both motors inoperative.

(c) Refer to Chapter 22 for autopilot pitch trim information.

HORIZONTAL STABILIZER CONTROL SYSTEM - MAINTENANCE PRACTICES

1. Adjustment/Test

NOTE: A master rigging plate and position indicator, located on the vertical stabilizer inner structure, is provided as an aid to check horizontal stabilizer position during maintenance. The rigging plate, marked to indicate allowable stabilizer travel tolerances, is to be used for reference only.

A. Rig Horizontal Stabilizer (See figure 201)

NOTE:
- On Aircraft 25-061 and 25-070 NOT modified per AAK 76-4 (Reduced Approach Speed System) or AMK 79-5B (Modification of Horizontal Stabilizer Trim Actuator), the upper screw is located 1° up from 0°. The lower screw is located 8°7' down from 0°.
- On Aircraft 25-061 and 25-070 thru 25-205 modified per AMK 79-5B (Modification of Horizontal Stabilizer Trim Actuator), the upper screw is located 0°15' down from 0°. The lower screw is located 8°7° down from 0°.
- On Aircraft 25-206 and Subsequent and prior Aircraft modified per AAK 76-4 (Reduced Approach Speed System), the upper screw is located 1° up from 0°. The lower screw is 9°0' down from 0°.

(1) Jack aircraft. (Refer to Chapter 7.)
(2) Level aircraft. (Refer to Chapter 8.)
(3) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.
(4) Remove fin tip upper forward fairing from vertical stabilizer.

NOTE: If the rigging plate has been removed or the pointer has been bent or broken and repaired, proceed with step (3). If the rigging plate and/or pointer has not been damaged, proceed to step (10).

(5) Drill out rivets which attach rigging plate to aircraft and remove rigging plate.
(6) Ensure that pointer is free to move through entire travel of horizontal stabilizer without interference or binding.
(7) Remove two countersunk screws at LBL 34.80 and install contour board mounting brackets and contour board (Tool No. 15785).
(8) Place vernier clinometer on level plate of contour board.
(9) Move stabilizer to 3°44' position. Scribe a mark extending path that pointer would make across rigging plate.
(10) Move stabilizer to full up position and extend mark scribed in step (7) to the right of where rigging plate is to be mounted.

(11) Position rigging plate so that 0° line on rigging plate aligns (vertically) with line scribed in step (8). Ensure that mounting holes of rigging plate align (horizontally) with existing mounting holes in aircraft structure. If necessary, mark and drill mounting holes. Rivet rigging plate in place.

(12) Set Pitch Trim Switch to NORM and check horizontal stabilizer for correct travel by moving either Pilot's or Copilot's Trim and Trim Arming Switch to NOSE UP, then NOSE DOWN.

NOTE: Stabilizer Travel Tolerance

- Aircraft 25-061 and 25-070 thru 25-205 NOT modified per AAK 76-4 (Reduced Approach Speed System) or AMK 79-5B (Modification of Horizontal Stabilizer Trim Actuator)
 Leading edge up from 0°; 1° to 1°30'.
 Leading edge down from 0°; 8°7' to 8°30°.
- Aircraft 25-061 and 25-070 thru 25-205 modified per AMK 79-5B (Modification of Horizontal Stabilizer Trim Actuator)
 Leading edge down from 0°; 0°15' to 0°30'.
 Leading edge down from 0°; 8°7' to 8°30'.
- Aircraft 25-206 and Subsequent and prior Aircraft modified per AAK 76-4 (Reduced Approach Speed System)
 Leading edge down from 0°; 0°30' to 1°0'.
 Leading edge down from 0°; 8°30' to 9°0'.

(13) Adjust actuator rod end as necessary. One complete turn of rod end changes actuator travel 0°10'.

(14) After stabilizer travel is within allowable tolerance, perform Inspection/Check.

(15) Install fin tip upper forward fairing and both horizontal stabilizer fairings.

2. Inspection/Check

A. Horizontal Stabilizer Pitch Trim Operational Check (Aircraft 25-061 and 25-070 thru 25-341, Except 25-337, not modified for Horizontal Stabilizer Trim and Autopilot Improvement)

(1) Connect a 28 vdc auxiliary power unit to aircraft.

(2) Set Battery Switch(es) and auxiliary power unit on.

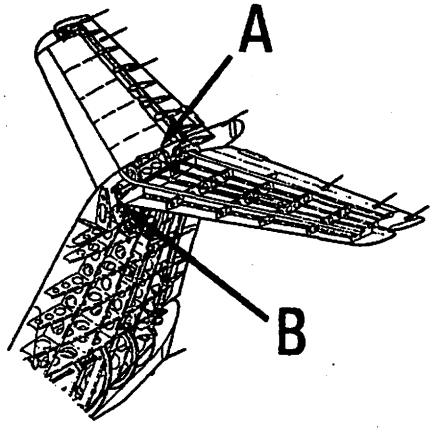
(3) Ensure that PITCH, EMER P TRIM, WRN LTS, FLAPS, AUD 1, and AFCS circuit breakers are depressed.

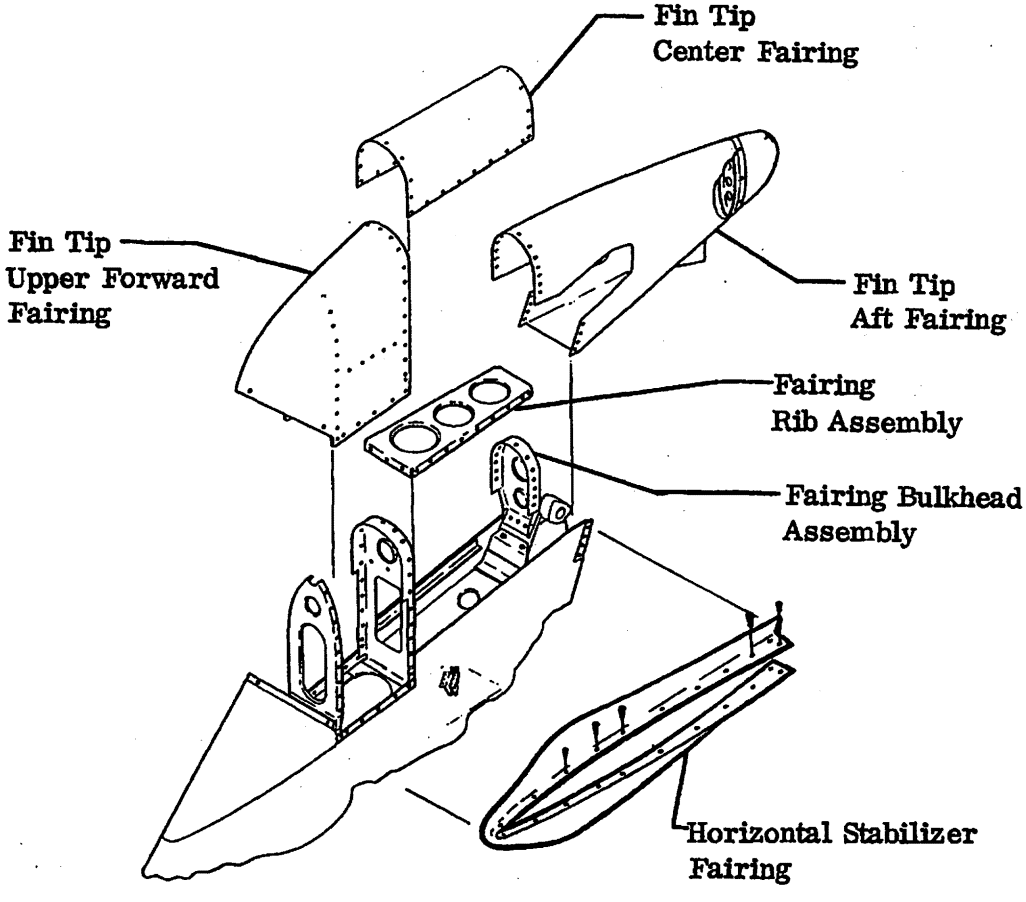
(4) Set Pitch Trim Switch to OFF.

(5) Actuate Pilot's and then Copilot's Trim and Trim Arming Switches. Horizontal stabilizer shall not move.

(6) Actuate Emergency Pitch Trim Switch. Horizontal stabilizer shall not move.

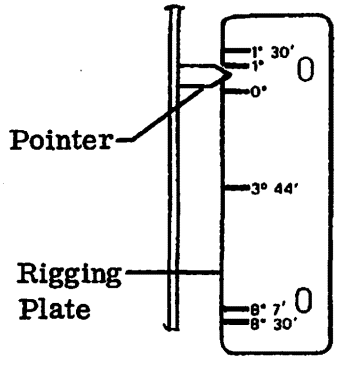
(7) Set Pitch Trim Switch to NORM.

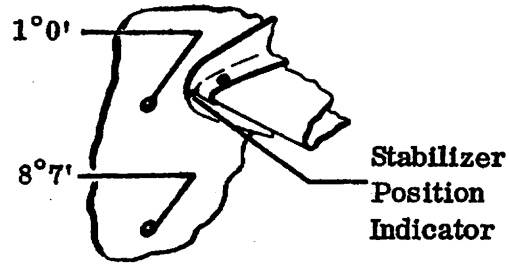




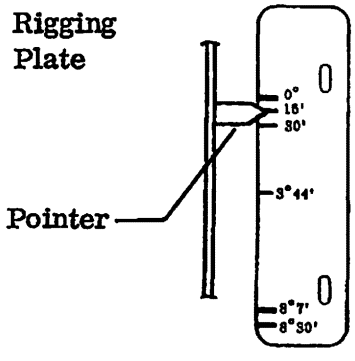
Detail A

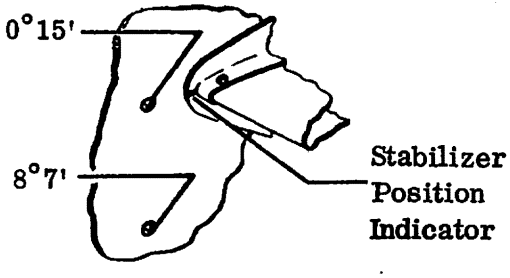
Horizontal Stabilizer Rigging
Figure 201 (Sheet 1 of 3)



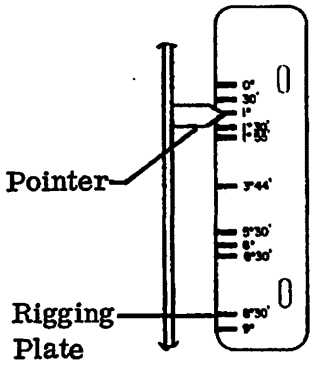


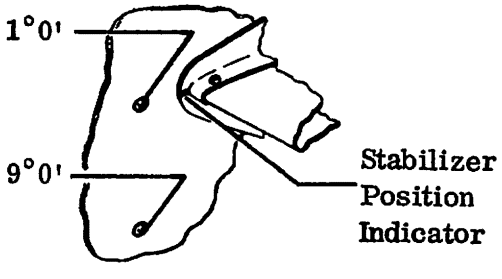
Aircraft 25-061 and 25-070 thru 25-205 Not modified per AAK 76-4 (Reduced Approach Speed System) or AMK 79-5 (Modification of Horizontal Stabilizer Trim Actuator)





Aircraft 25-061 and 25-070 thru 25-205 modified per AMK79-5 (Modification of Horizontal Stabilizer Trim Actuator)



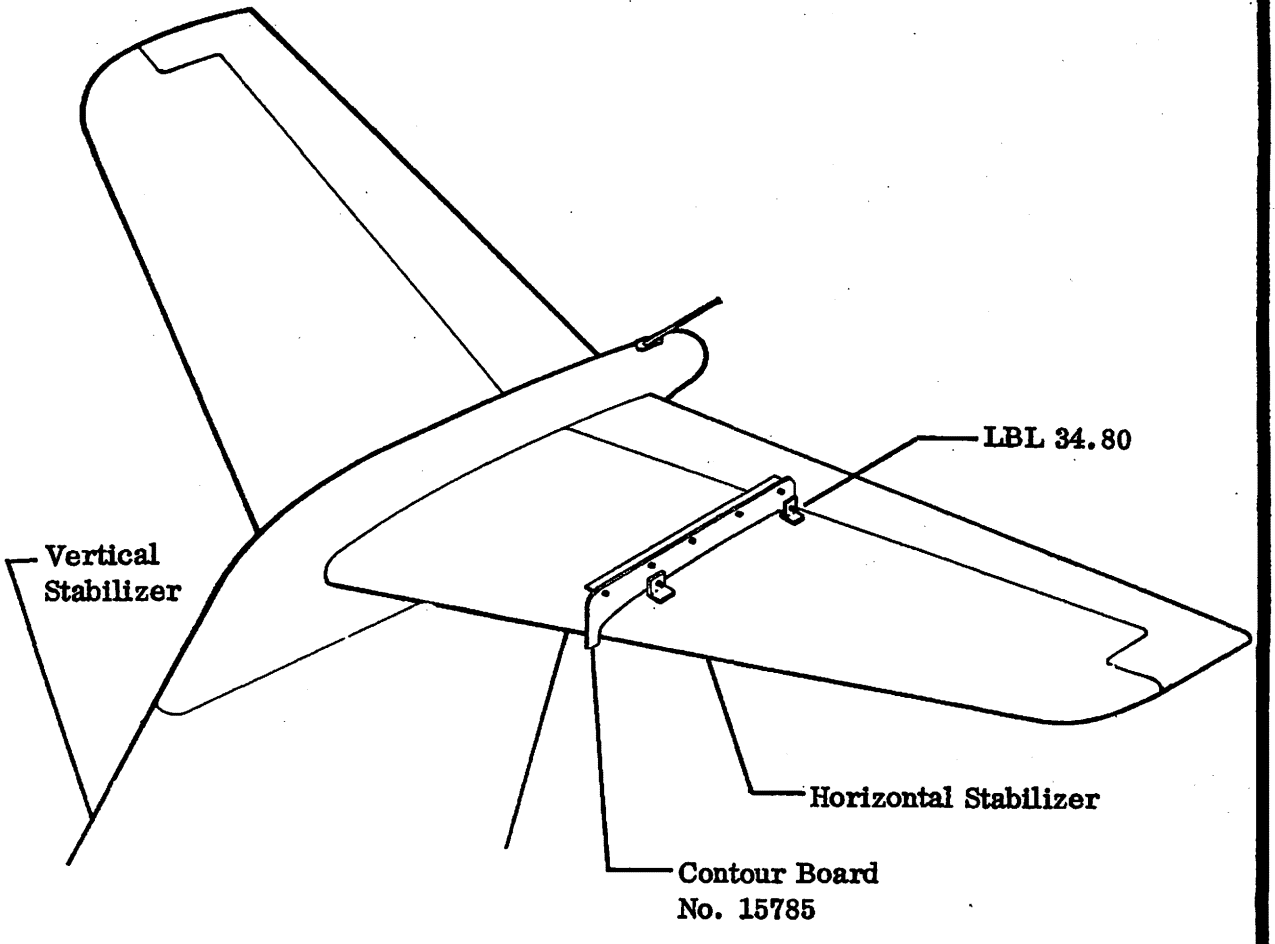


Aircraft 25-206 and Subsequent and prior Aircraft modified per AAK 76-4 (Reduced Approach Speed System)

Detail B

Horizontal Stabilizer Rigging
Figure 201 (Sheet 2 of 3)

8-83A



Horizontal Stabilizer Rigging
Figure 201 (Sheet 3 of 3)

(8) Actuate Pilot's and then Copilot's Trim and Trim Arming Switch. Horizontal stabilizer movement shall correspond to switch position.

(9) Actuate Pilot and Copilot's Trim and Trim Arming Switches at same time, commanding opposite directions. Horizontal stabilizer movement shall correspond to Pilot's Trim and Trim Arming Switch position.

(10) Actuate Pilot's and then Copilot's Trim and Trim Arming Switch with Emergency Pitch Trim Switch set to NOSE DOWN. Repeat this procedure with Emergency Pitch Trim Switch set to NOSE UP. Horizontal stabilizer movement shall correspond to Trim and Trim Arming Switch position regardless of Emergency Pitch Trim Switch position.

(12) Set Emergency Pitch Trim Switch to NOSE UP, then NOSE DOWN. Horizontal stabilizer shall not move.

(13) Set Pitch Trim Switch to EMER.

(14) Actuate Pilot's and then Copilot's Trim and Trim Arming Switch. Horizontal stabilizer shall not move.

(15) Actuate Emergency Pitch Trim Switch. Horizontal stabilizer movement shall correspond to switch position.

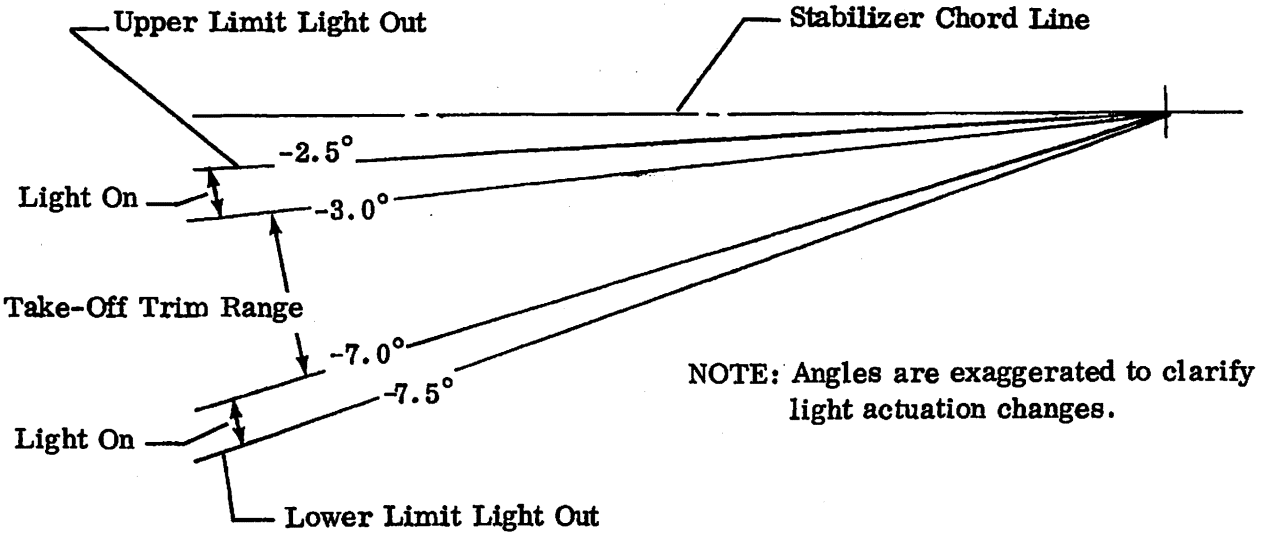
(16) Repeat step (15) and depress and release Pilot's Master Switch, then Copilot's Master Switch during horizontal stabilizer movement. Movement shal not stop when either Master Switch is depressed.

(17) Disconnect 28 vdc power from aircraft.

B. On Aircraft 25-337, 25-342 and Subsequent and Prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement, refer to 27-00-00 for Trim Systems Operational Check.

C. Take-Off-Trim Operational Check (Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement) (See figure 202)

(1) Attach a contour board and inclinometer to horizontal stabilizer.
(2) Connect a 28 vdc power supply to aircraft and set Battery Switches on.
(3) Using either pilot's or copilot's Trim and Trim Arming Switch, position stabilizer to -2.5° to -3.0° leading edge down. TAKE-OFF-TRIM warning light shall be illuminated.
(4) Position stabilizer at -7° to -7.5° leading edge down. TAKE-OFF-TRIM warning light shall be illuminated.
(5) Drive stabilizer toward -2.5° until TAKE-OFF-TRIM warning light just energizes. Note actual stabilizer position.
(6) Drive stabilizer toward -7.5° position until light just energizes. Note actual stabilizer position.
(7) Should stabilizer position not fall within tolerances given in steps (3) and (4) and figure 202, refer to Overhaul Manual 1711-9 for adjustment procedures.



Take-Off-Trim Angles
Figure 202

HORIZONTAL STABILIZER - MAINTENANCE PRACTICES

1. Removal/Installation

A. Tools and Equipment

NAME	NUMBER	MANUFACTURER	USE
Hoist & Sling	2331001	Gates Learjet Corp.	Support stabilizer.
Driver	MIT 2331028	Gates Learjet Corp.	Remove hinge pin.

B. Remove Horizontal Stabilizer (See figure 201)

(1) Set Battery Switch(es) to on position.

(2) Using Pilot's Trim and Trim Arming Switch, depress and set to NOSE UP and hold. When horizontal stabilizer has reached extreme down position, release Trim and Trim Arming Switch.

(3) Set Battery Switch(es) to OFF.

(4) Cage elevator downspring. (Refer to 27-30-01.)

(5) Remove elevators. (Refer to 27-30-00.)

(6) Remove fin tip upper forward and center fairings, fairing rib assembly, and fairing bulkhead assembly from vertical stabilizer. Remove hinge pin access covers from RH and LH sides of horizontal stabilizer. Leave opposite end connected to vertical stabilizer hinge fittings.

(7) Disconnect horizontal stabilizer anti-ice wiring. (Refer to Chapter 30.)

(8) Remove attaching parts and elevator downspring push-pull tube from horizontal stabilizer.

(9) Remove screws from upper surface of horizontal stabilizer. The screws are located outboard of BL 19 on both sides. Attach hoist and support sling (Learjet P/N 2331001) to horizontal stabilizer using screws (P/N NAS220-15).

(10) Remove attaching parts from horizontal stabilizer actuator and stabilizer attach point.

(11) Remove safety wire, bolt, and stop from hinge pin.

(12) Remove hinge pin using 5X rivet gun and driver (P/N MIT 2331028).

(13) Remove horizontal stabilizer from aircraft. Use care when removing horizontal stabilizer not to damage rigging plate pointer.

C. Install Horizontal Stabilizer (See figure 201)

(1) Clean hinge pin and place in dry ice for a minimum of 4 hours prior to installation.

(2) Position horizontal stabilizer so that holes in vertical stabilizer fittings and horizontal stabilizer hinge fittings are aligned.

(3) Remove hinge pin from dry ice container and insert in aligned holes. Drive hinge pin into place (head of hinge pin flush with stabilizer fitting) using 5X rivet gun and driver (P/N MIT 2331028). Hinge pin must be driven in place within a maximum of 14 seconds after removal

from dry ice.

(4) Install stop, washer, and bolt. Check for clearance between stop prongs and hinge fitting as shown in detail C. Tighten bolt and safety wire.

(5) Secure bonding jumper to stabilizer with attaching parts.

(6) Secure stabilizer actuator to stabilizer with attaching parts. (Refer to 27-40-02.)

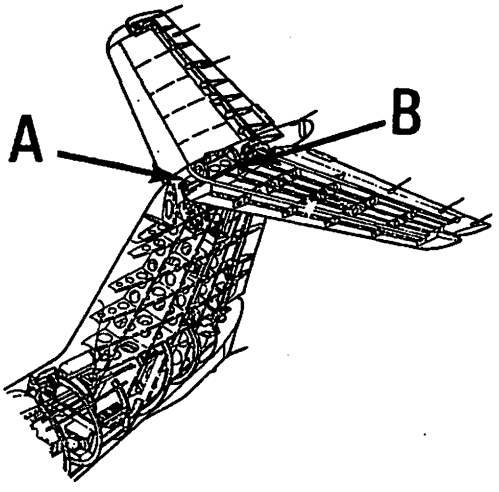
(7) Install elevators. (Refer to 27-30-00.) Do not install fairings at this time.

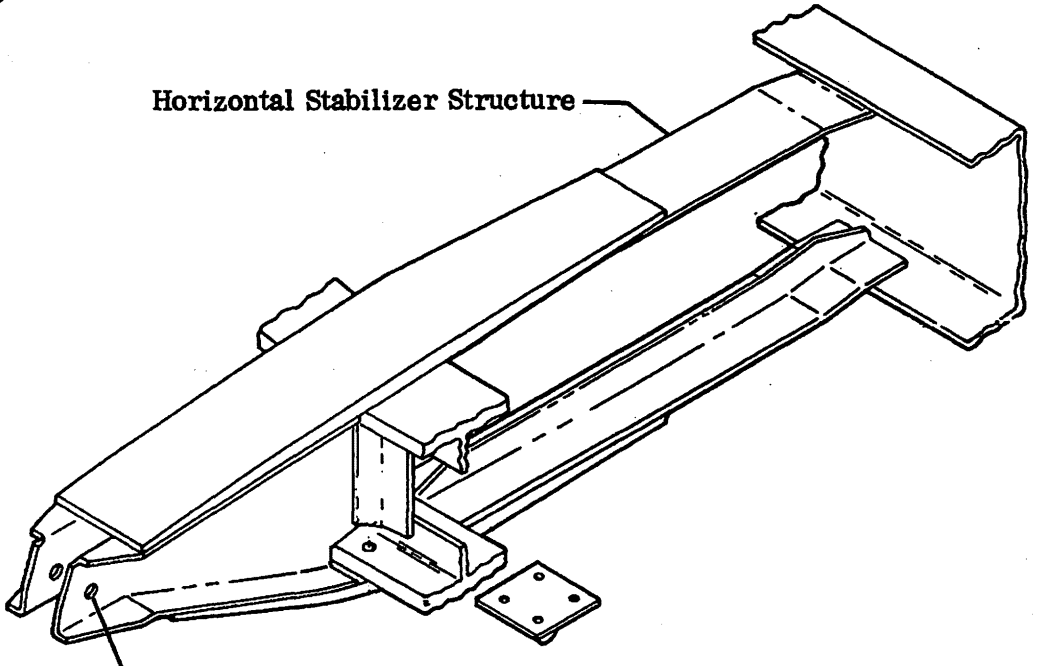
(8) Remove elevator downspring caging pin. (Refer to 27-30-01.)

(9) Connect stabilizer anti-ice wiring. (Refer to Chapter 30.)

(10) Install fairing bulkhead assembly, fairing rib assembly, fin tip upper forward, center, and aft fairings, horizontal stabilizer fairings, and hinge pin access covers.

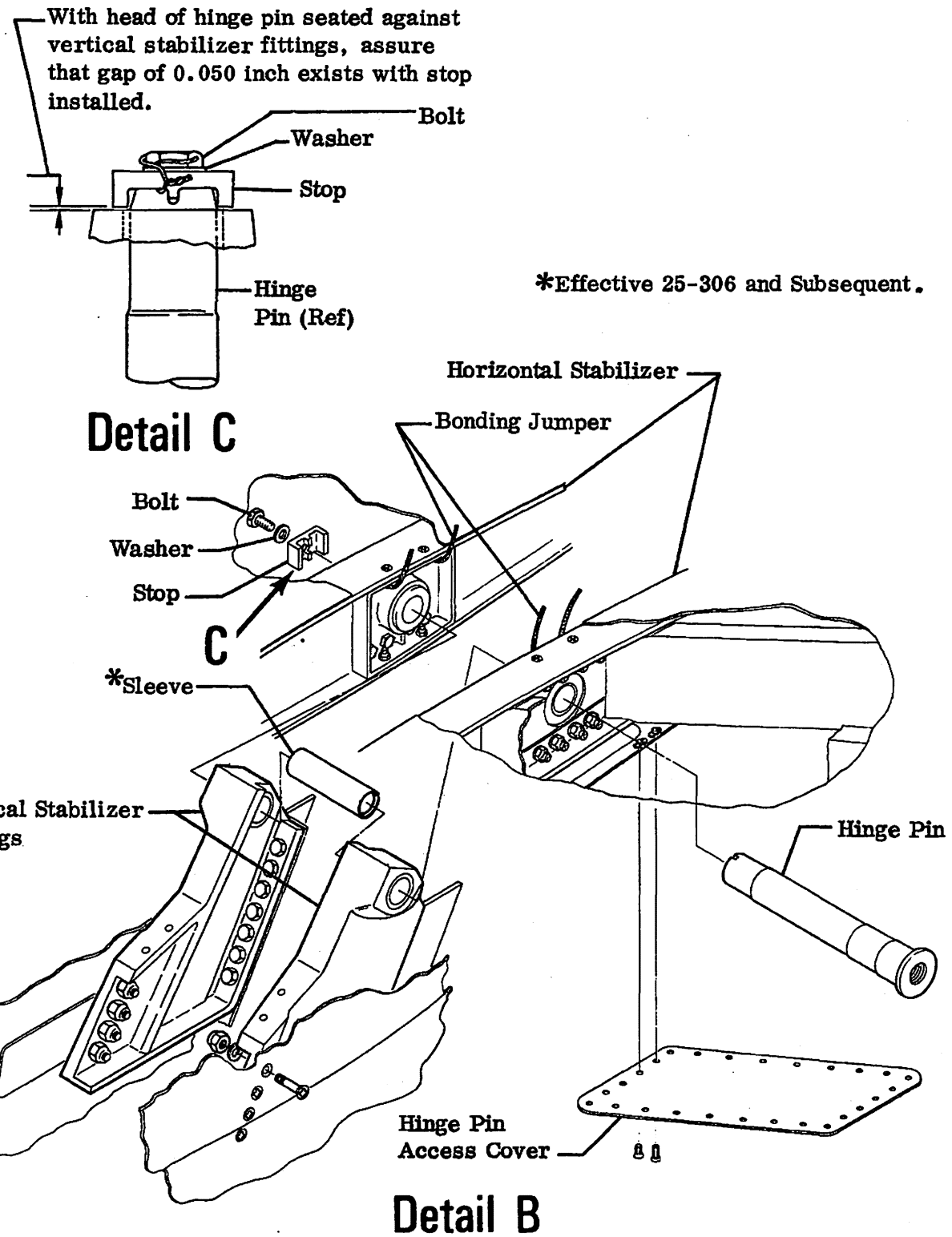
(11) Check clearances of fairings. (Refer to 27-30-00 for clearances.)





Detail A

Horizontal Stabilizer Installation
Figure 201 (Sheet 1 of 2)



Horizontal Stabilizer Installation
Figure 201 (Sheet 2 of 2)

8-7C

HORIZONTAL STABILIZER ACTUATOR - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Horizontal Stabilizer Actuator Aircraft 25-061, 25-070 thru 25-199. (See figure 201)

(1) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.

(2) Remove fin tip upper forward fairing and stabilizer actuator access door from vertical stabilizer.

(3) Disconnect electrical connector from actuator.

(4) Remove attaching parts securing actuator to horizontal stabilizer.

(5) On Aircraft 25-194 thru 25-199 and prior Aircraft modified per AAK 76-4 (Reduced Approach Speed System), remove attaching parts securing actuator ground strap.

(6) Remove attaching parts securing actuator to vertical stabilizer spar. Remove actuator from aircraft.

B. Install Horizontal Stabilizer Actuator Aircraft 25-061, 25-070 thru 25-199. (See figure 201)

(1) Attach actuator lower fitting to vertical stabilizer spar and secure with attaching parts.

(2) On Aircraft 25-194 thru 25-199 and prior Aircraft modified per AAK 76-4 (Reduced Approach Speed System), secure actuator ground strap to vertical stabilizer spar with attaching parts.

(3) Attach actuator upper fitting to horizontal stabilizer and secure with attaching parts.

(4) Connect electrical connector to actuator.

(5) Check actuator for proper operation and horizontal stabilizer for proper travel. (Refer to 27-40-00, Adjustment/Test.)

(6) Install fin tip upper forward fairing and stabilizer actuator access door on vertical stabilizer.

(7) Install horizontal stabilizer fairings on both sides of vertical stabilizer.

(8) Check clearance (0.31 (±0.12) inch) between horizontal stabilizer fairing and vertical stabilizer.

C. Remove Horizontal Stabilizer Actuator Aircraft 25-200 and Subsequent (See figure 201)

(1) Remove horizontal stabilizer fairings from both sides of vertical stabilizer.

(2) Remove fin tip upper forward fairing and stabilizer actuator access door from vertical stabilizer.

(3) Disconnect electrical connectors from actuator.

(4) Remove attaching parts and upper actuator bolt retainer.

(5) Remove attaching parts securing actuator to horizontal stabilizer.

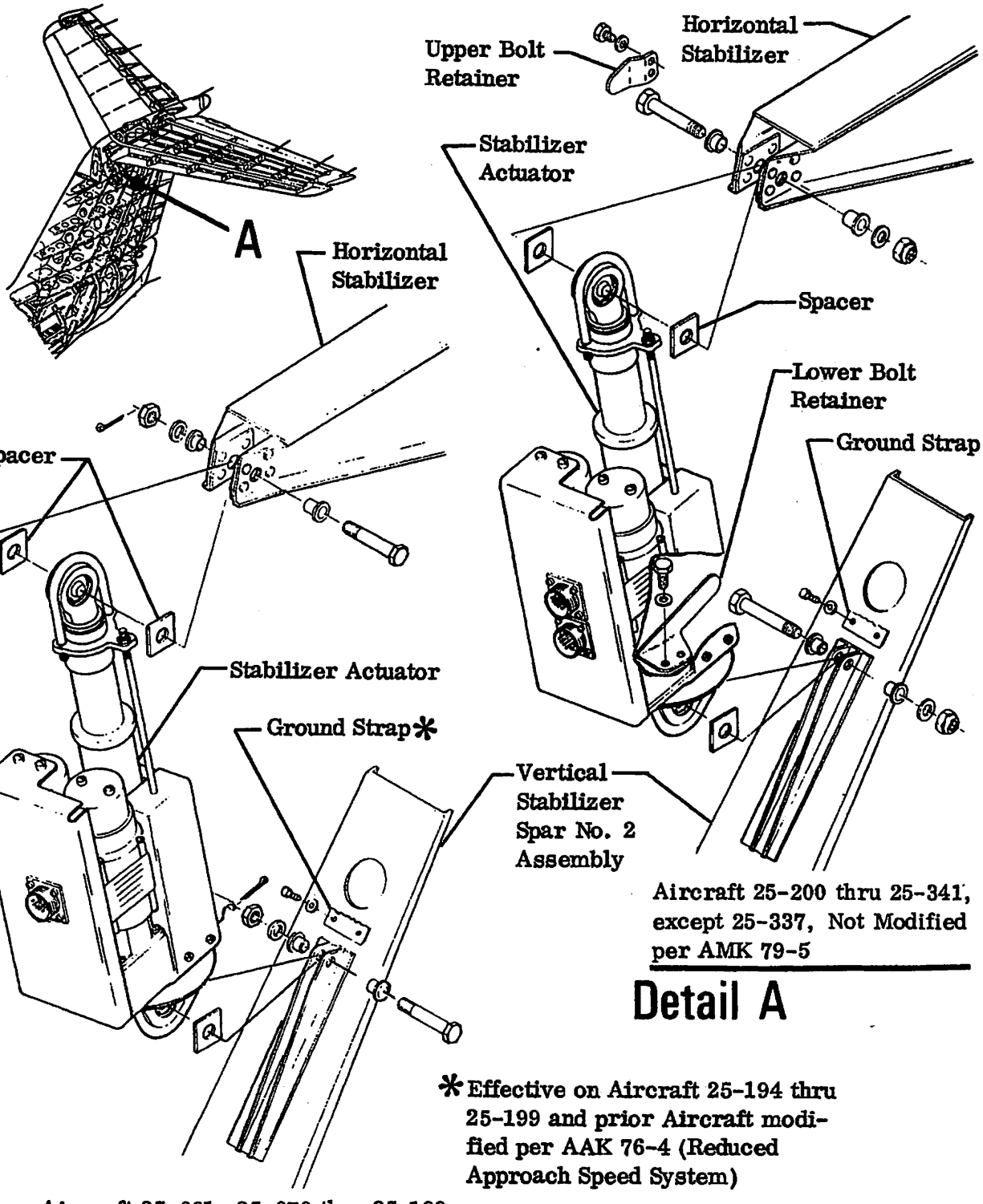
(6) Remove attaching parts and lower actuator bolt retainer.

(7) Remove attaching parts securing actuator ground strap.

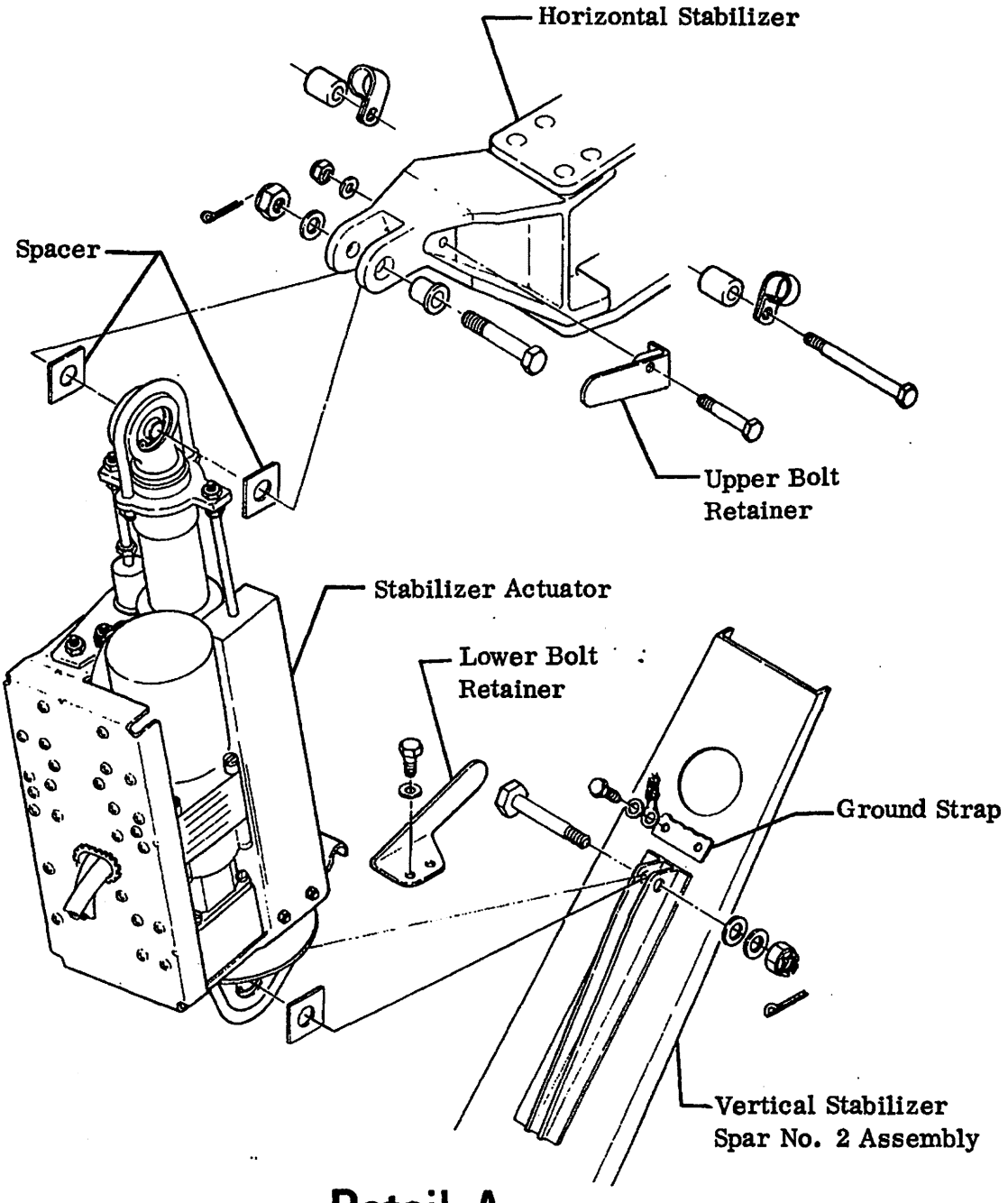
(8) Remove attaching parts securing actuator to vertical stabilizer spar. Remove actuator from aircraft.

D. Install Horizontal Stabilizer Actuator Aircraft 25-200 and Subsequent (See figure 201)

(1) Install actuator and secure lower fitting to vertical stabilizer spar with attaching parts.
(2) Install lower bolt retainer and secure with attaching parts.
(3) Secure actuator ground strap to vertical stabilizer spar with attaching parts.
(4) Secure upper actuator fitting to horizontal stabilizer with attaching parts.
(5) Install upper bolt retainer and secure with attaching parts.
(6) Connect electrical connectors to actuator.
(7) Check actuator for proper operation and horizontal stabilizer for proper travel. (Refer to 27-40-00, Adjustment/Test.)
(8) Install fin tip upper forward fairing and stabilizer actuator access door on vertical stabilizer.
(9) Install horizontal stabilizer fairing on both sides of vertical stabilizer.
(10) Check clearances of fairing. (Refer to 27-30-00 for clearances.)



Horizontal Stabilizer Actuator Installation
Figure 201 (Sheet 1 of 2)



Detail A

Horizontal Stabilizer Actuator Installation
Figure 201 (Sheet 2 of 2)

8-8C-3

STABILIZER TRIM POSITION INDICATOR - MAINTENANCE PRACTICES

1. Removal/Installation
 A. Remove Trim Position Indicator (See figure 201)

 NOTE: The trim position indicators are located on the aft end of the pedestal.

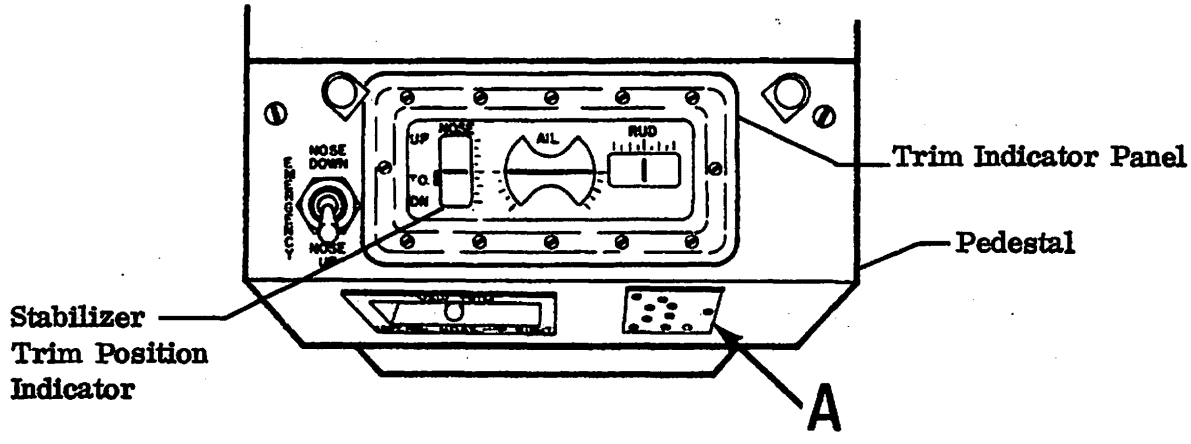
 (1) Remove trim indicator panel attaching parts.
 (2) Remove indicator panel from pedestal assembly far enough to disconnect electrical connector from stabilizer trim position indicator.
 (3) Remove attaching parts and indicator from panel.

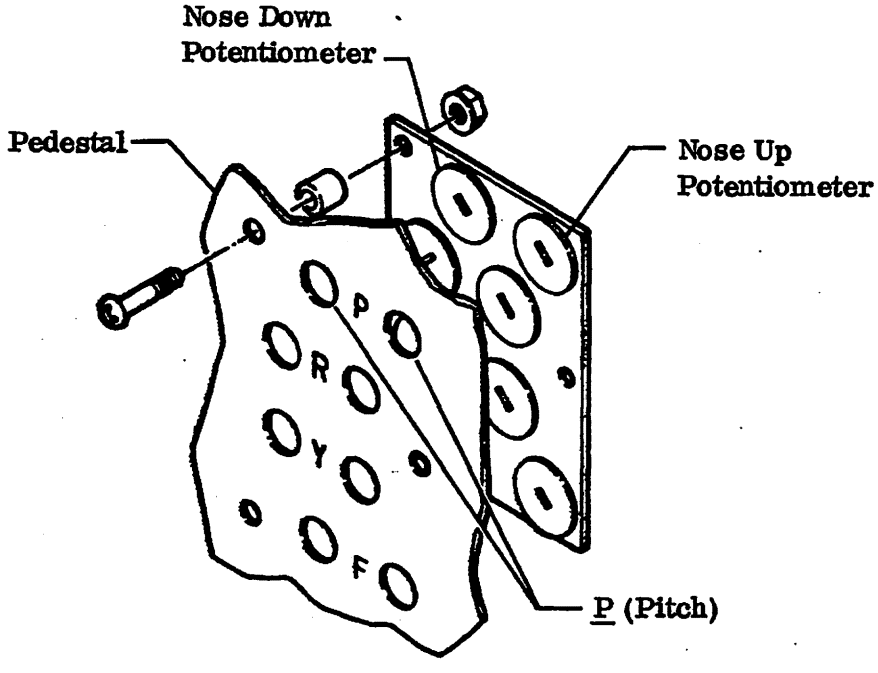
 B. Install Trim Position Indicator (See figure 201)
 (1) Position stabilizer trim position indicator in trim indicator panel and secure with attaching parts.
 (2) Connect electrical connector to indicator.
 (3) Install trim indicator panel in pedestal assembly and secure with attaching parts.

2. Adjustment/Test
 A. Calibrate Stabilizer Trim Position Indicator (See figure 201)
 (1) Actuate horizontal stabilizer to full down position.
 (2) Adjust nose up potentiometer to obtain full NOSE UP indication on stabilizer trim position indicator.

 NOTE: The position indicator calibration assembly is located on the aft side of the pedestal.

 (3) Actuate horizontal stabilizer to full up position.
 (4) Adjust nose down potentiometer to obtain full NOSE DN indication on stabilizer trim position indicator.
 (5) Cycle horizontal stabilizer between full up and full down positions while checking respective indications on pitch trim indicator. Readjust nose up and nose down potentiometers as necessary to obtain coordinated indications.





Detail A

Position Indicator Calibration Assembly
Figure 201

8-107A

PITCH TRIM SPEED CONTROLLER - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Pitch Trim Speed Controller (See figure 201)

(1) Ensure that Battery Switches are set to OFF and that external power is disconnected.

(2) Remove vertical stabilizer access panel located on RH side between WL 85 and WL 95 and between spars 1 and 2 to gain access to pitch trim speed controller.

(3) Disconnect electrical connectors from pitch trim speed controller.

(4) Remove attaching parts and pitch trim speed controller assembly from aircraft.

B. Install Pitch Trim Speed Controller (See figure 201)

(1) Position pitch trim speed controller in vertical stabilizer and secure with attaching parts.

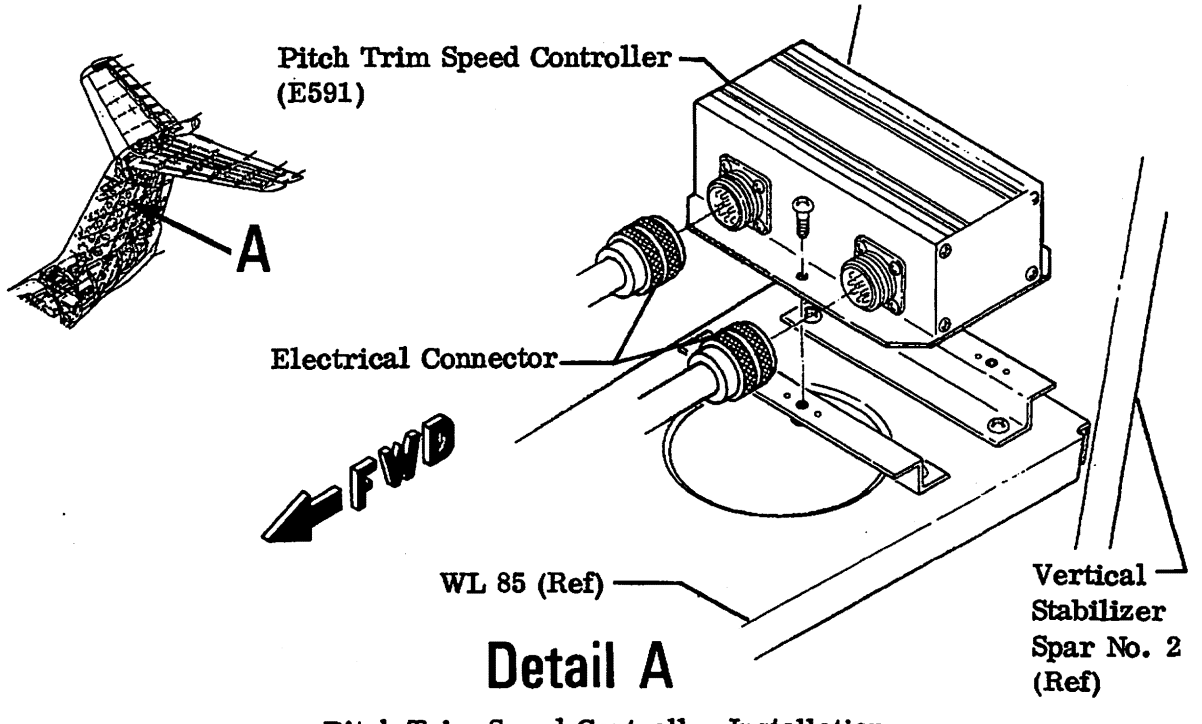
NOTE: Install bonding jumper between pitch trim speed controller box frame and aircraft structure. Check electrical resistance as instructed in Chapter 20 of the Wiring Manual.

(2) Connect electrical connectors to pitch trim speed controller.

(3) Install and secure vertical stabilizer access panel.

(4) Restore electrical power to aircraft.

(5) Perform Horizontal Stabilizer Pitch Trim Operational Check.



Pitch Trim Speed Controller Installation
Figure 201

2. Inspection/Check

A. Pitch Trim Speed Controller Operational Check

(1) Apply 28 vdc auxiliary power to aircraft.
(2) Set Battery Switches on.
(3) Set Pitch Trim Switch to NORM.
(4) Position flaps up.
(5) Measure elapsed time as stabilizer moves from up stop to down stop using either pilot's or copilot's Trim and Trim Arming Switch.
(6) Stop to stop elapsed time shall be 75 (+3/-0) seconds.
(7) Position flaps down.
(8) Measure elapsed time as stabilizer moves from up stop to down stop using either Pilot's or Copilot's Trim and Trim Arming Switch.
(9) Stop to stop elapsed time shall be 16 (+4/-0) seconds.
(10) Set Pitch Trim Switch to EMER.
(11) Actuate Emergency Pitch Trim Switch while measuring elapsed time from stop to stop.
(12) Stop to stop elapsed time shall be 51 (+24/-0) seconds.
(13) Operate trim from either pilot's or copilot's side while raising flaps simultaneously. Trim speed shall change from high speed to low speed as flaps retract through 3°.
(14) Position flaps down.
(15) Operate trim from either pilot's or copilot's side while depressing Pitch Trim Monitor Test Switch. Trim warning light shall illuminate. Raise flaps; trim warning light shall extinguish.

PITCH TRIM DISABLE TERMINAL BOARD - MAINTENANCE PRACTICES

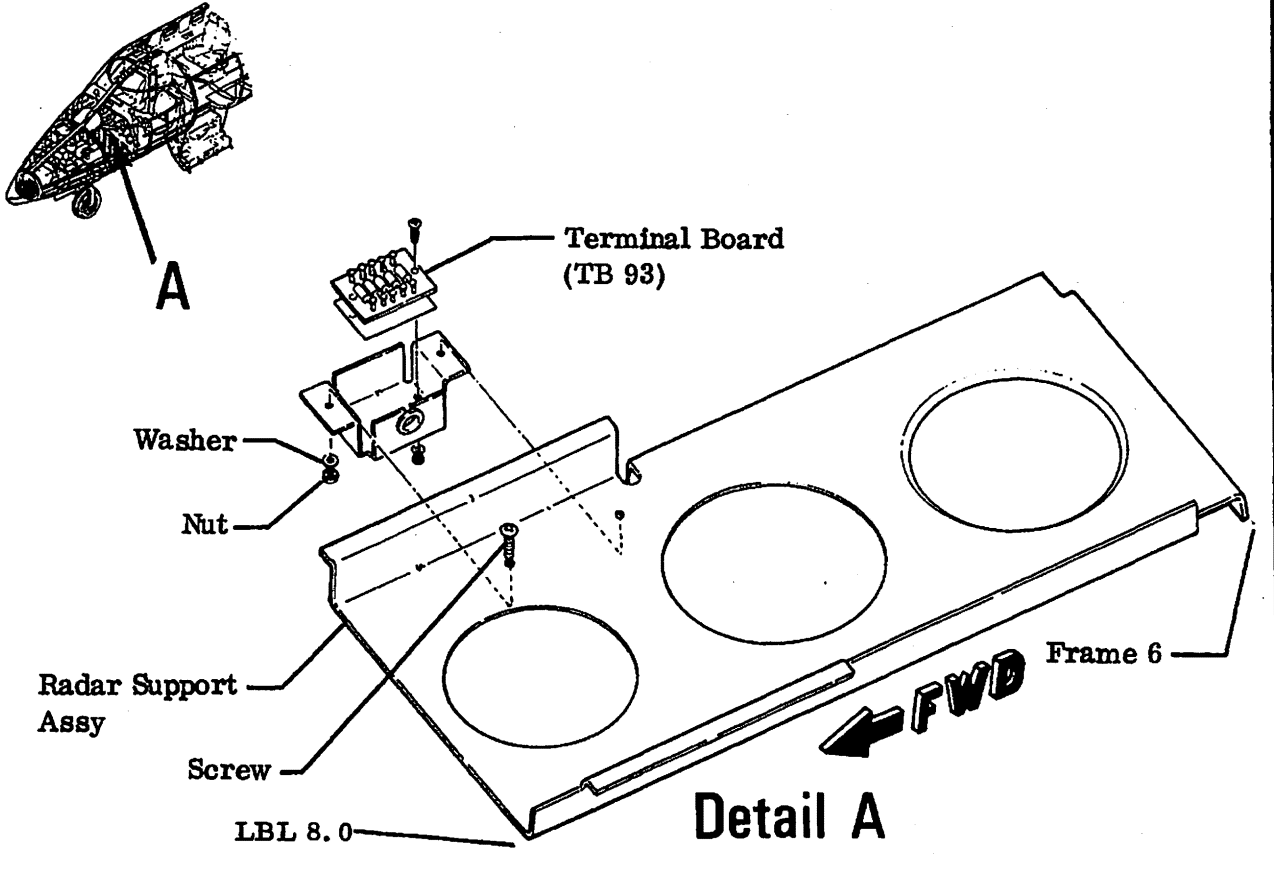
1. Removal/Installation

A. Remove Pitch Trim Disable Terminal Board (See figure 201)

(1) Remove electrical power from aircraft.
(2) Remove radar unit to gain access to top of radar support assembly.
(3) Disconnect and tag electrical wiring from terminal board.
(4) Remove attaching parts and terminal board and cover assembly from aircraft.
(5) Remove attaching parts and terminal board from cover.

B. Install Pitch Trim Disable Terminal Board (See figure 201)

(1) Position terminal board in cover and secure with attaching parts.
(2) Position terminal board and cover assembly on bottom of radar support assembly.
(3) Connect electrical wiring to terminal board.
(4) Install radar unit.
(5) Restore power to aircraft.
(6) Perform Horizontal Stabilizer Pitch Trim Operational Check. Refer to 27-40-00.

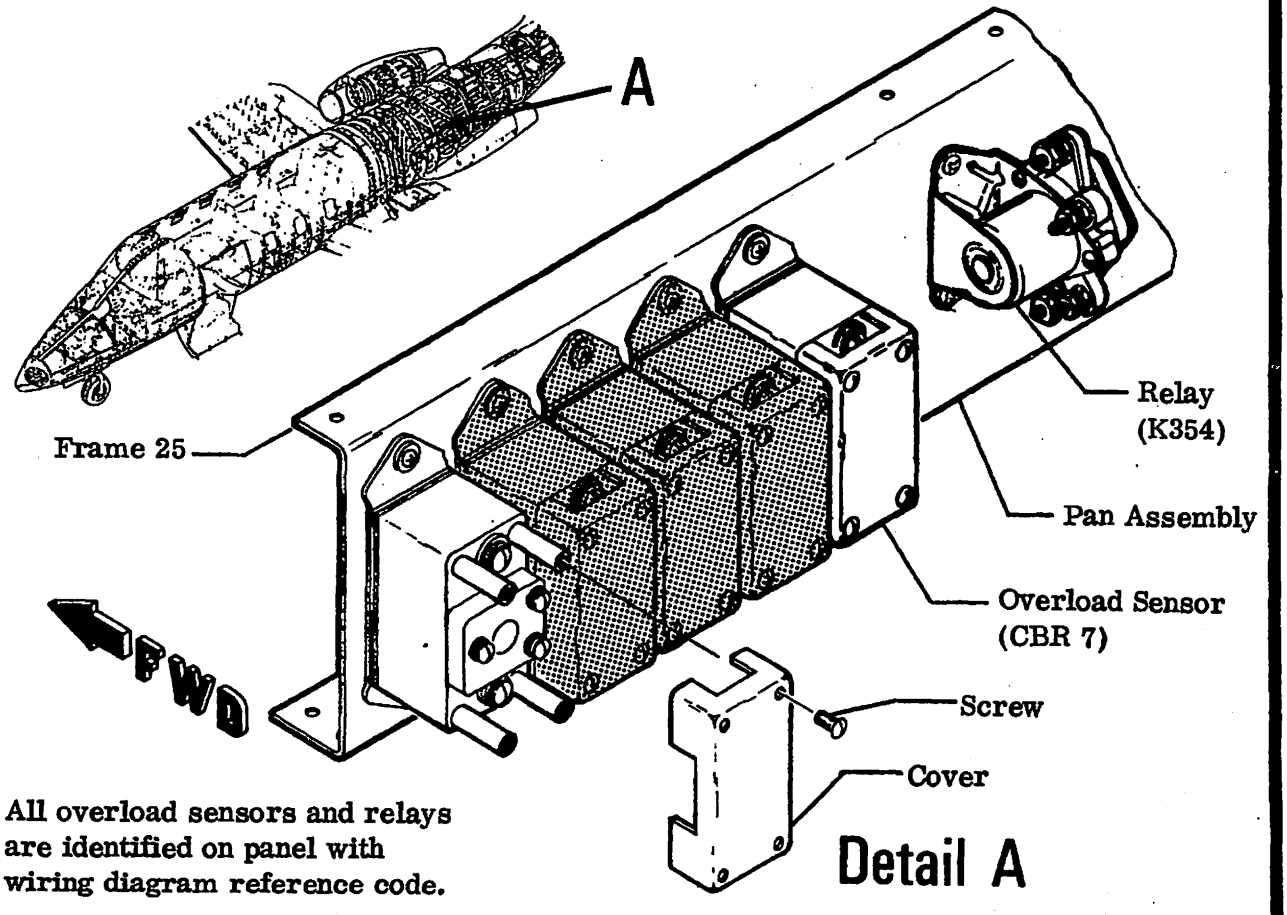


Pitch Trim Disable Terminal Board Installation
Figure 201

8-109B

HORIZONTAL STABILIZER ACTUATOR OVERLOAD SENSOR AND RELAY - MAINTENANCE PRACTICES

1. Removal/Installation
 - A. Remove Overload Sensor (See figure 201)
 - (1) Open tailcone access door and remove electrical power from aircraft.
 - (2) Remove attaching parts and overload sensor cover.
 - (3) Disconnect electrical wiring from overload sensor. Tag electrical wiring.
 - (4) Remove attaching parts and overload sensor from aircraft.
 - B. Install Overload Sensor (See figure 201)
 - (1) Install overload sensor and secure with attaching parts.
 - (2) Remove tags and connect electrical wiring to overload sensor. Install cover and secure with attaching parts.
 - (3) Restore electrical power to aircraft.
 - (4) Perform Horizontal Stabilizer Pitch Trim Operational Check. (Refer to 27-40-00.)
 - (5) Close tailcone access door.



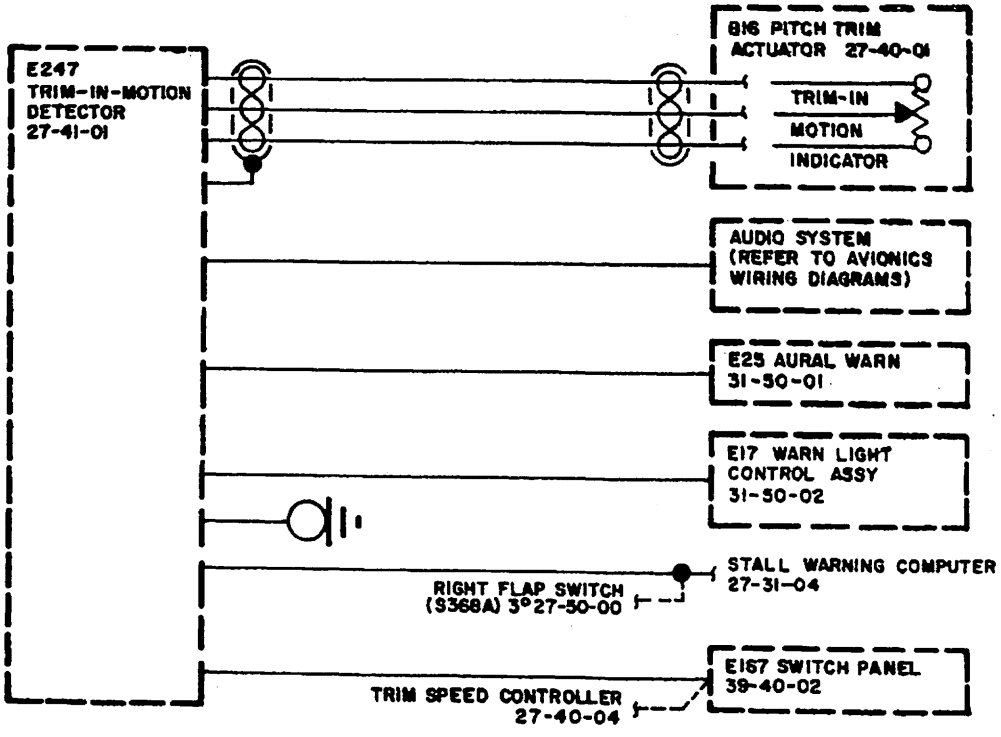
Overload Sensor and Relay Installation
Figure 201

9-45C

TRIM-IN-MOTION SYSTEM - DESCRIPTION AND OPERATION

1. Description (See figure 1)
 A. The trim-in-motion system alerts the crew to pitch trim motion (with flaps up) by sounding a warning (clicking) through the cockpit speaker and headphone audio systems.
 B. The trim-in-motion system components are a trim-in-motion detector box located forward of FS 235 (frame 10) between stringers 14 and 15, a linear potentiometer mounted on the horizontal stabilizer actuator, and a 3° flap switch.

2. Operation (See figure 1)
 A. Horizontal stabilizer actuator movement (pitch trim) modulates signal output of the actuator linear potentiometer which acts on the trim-in-motion detector box circuitry. The trim-in-motion detector box signals the audible tone generator which produces a series of clicks heard through the cockpit speakers and headphones. Sound start is delayed to begin 1 (±0.2) seconds after trim command, allowing the crew to pulse trim the aircraft without audible warning.
 B. The trim-in-motion system (aural warning) is inoperative when the flaps are extended beyond 3°.



Trim-in-Motion System Electrical Schematic
Figure 1

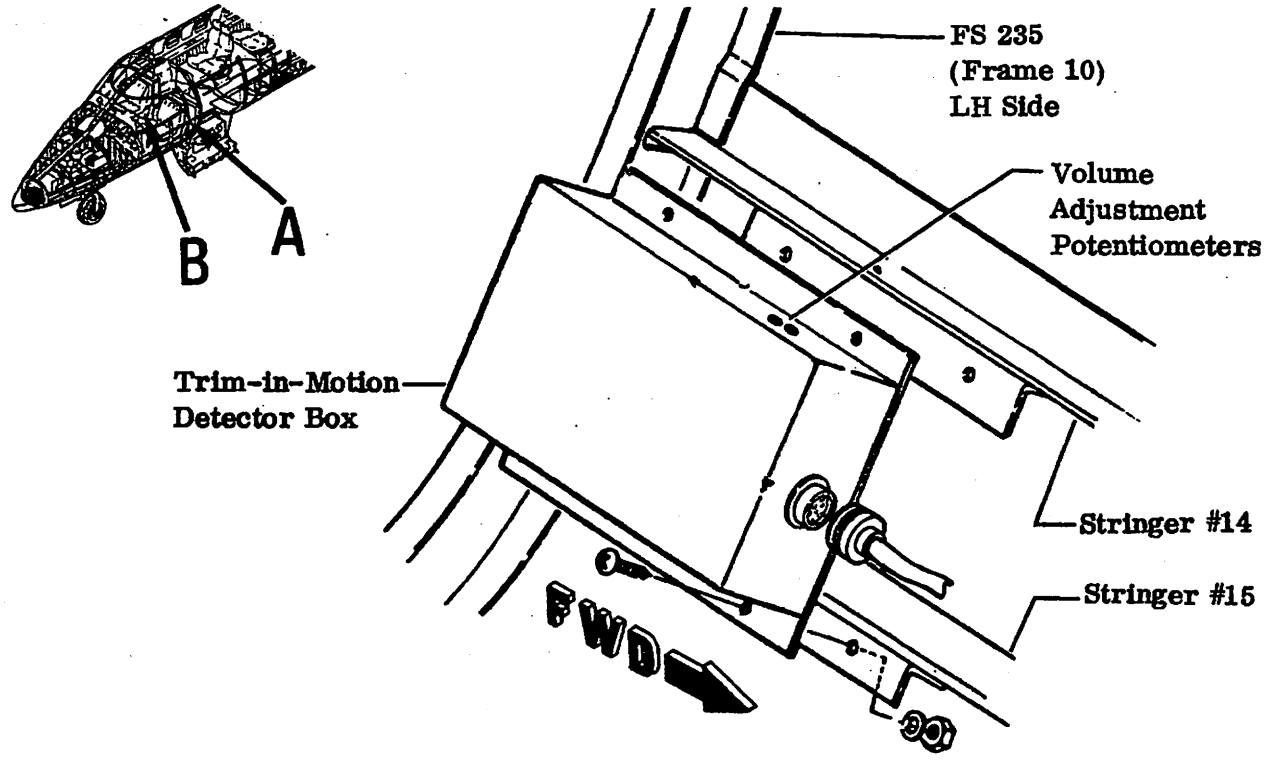
TRIM-IN-MOTION SYSTEM - MAINTENANCE PRACTICES

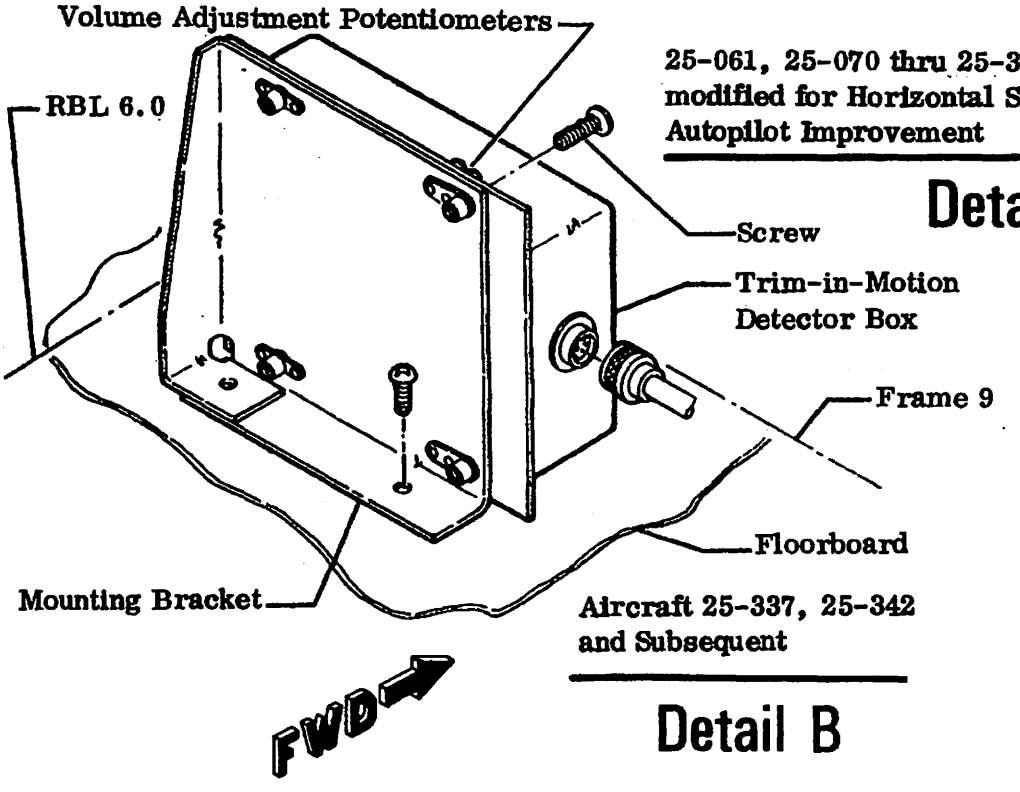
1. Adjustment/Test
 A. Trim-in-Motion System Volume Adjustment (See figure 201.)

 NOTE: It is recommended that volume adjustment be accomplished during a high altitude, high-speed cruise to assure audibility at maximum normal noise levels.

 (1) Ensure that flaps are fully retracted.
 (2) Activate either the primary or secondary pitch trim systems.
 (3) Operate pitch trim and adjust potentiometer on trim-in-motion detector box to vary volume.

2. Inspection/Check
 A. Trim-in-Motion System Operational Check
 (1) Assure that flaps are fully retracted.
 (2) Activate primary pitch trim system by setting Pitch Trim Switch to NORM and either Pilot's or Copilot's Trim and Trim Arming Switch to NOSE UP or NOSE DOWN (arming button depressed). Audio clicking (cricket) shall sound at (±0.5) second after stabilizer movement begins and shall silence 0.2 second after stabilizer movement stops.
 (3) Activate secondary pitch trim system by setting Pitch Trim Switch to EMER and Emergency Pitch Trim Switch to NOSE UP or NOSE DOWN. Note trim-in-motion audio as in step (2).
 (4) Set Pitch Trim Switch to NORM. Using Pilot's Control Wheel Trim Switch (arming button depressed), actuate horizontal stabilizer NOSE UP and NOSE DOWN. While trimming, depress either Control Wheel Master Switch; audio clicking (cricket) shall stop. Pitch trim cannot be reactivated until the arming button and the Control Wheel Master Switch are released.
 (5) Perform steps (2) through (4) with flaps extended to 3°. Trim-in-motion audio shall not sound.
 (6) Perform steps (2) through (4) with flaps extended to 13°. Trim-in-motion audio shall not sound.





25-061, 25-070 thru 25-341, except 25-337, modified for Horizontal Stabilizer Trim and Autopilot Improvement

Detail A

Aircraft 25-337, 25-342 and Subsequent

Detail B

Trim-in-Motion System Volume Adjustment Potentiometer Installation
Figure 201

8-96B

TRIM-IN-MOTION DETECTOR BOX - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Trim-in-Motion Detector Box (See figure 202)

(1) Remove electrical power from aircraft.

(2) On Aircraft 25-061, 25-070 thru 25-341, except 25-337, modified for Horizontal Stabilizer Trim and Autopilot Improvement, locate trim-in-motion detector box on LH side of cabin 2 inches forward of frame 10 between stringers 14 and 15. Remove upholstery as required.

(3) On Aircraft 25-337, 25-342 and Subsequent, locate trim-in-motion detector box on floorboard behind copilot's seat.

(4) Disconnect electrical connector from detector box.

(5) Remove attaching parts and detector box from aircraft.

B. Install Trim-in-Motion Detector Box (See figure 201)

(1) Position trim-in-motion detector box and secure with attaching parts.

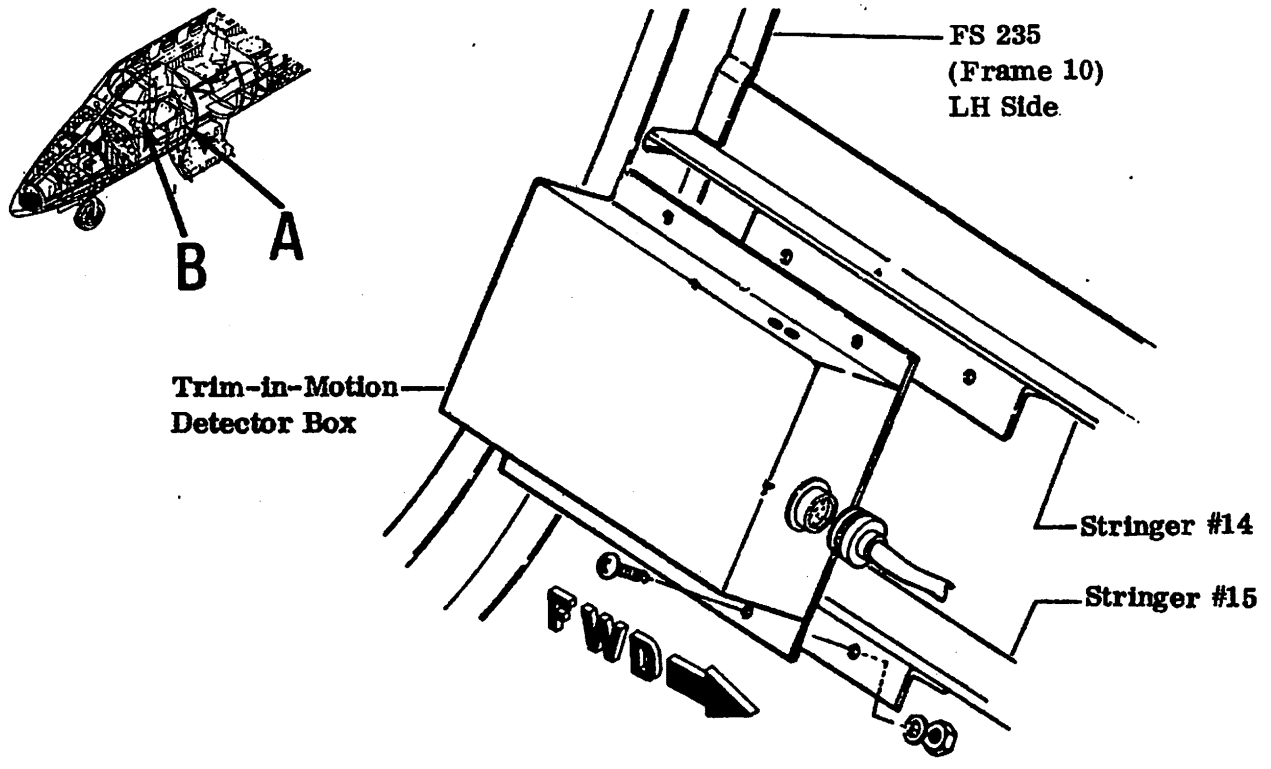
(2) Connect electrical connector to detector box.

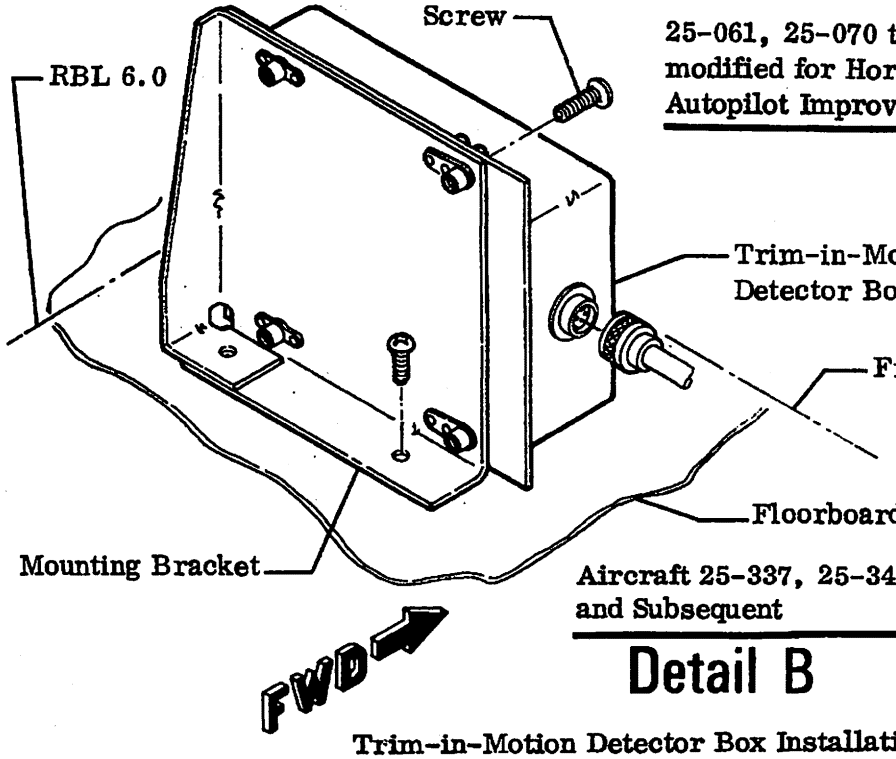
(3) Install any previously removed upholstery.

(4) Restore power to aircraft.

(5) Perform Trim-in-Motion System Operational Check. (Refer to Inspection/Check, 27-41-00.)

(6) Perform Trim-in-Motion System Volume Adjustment. (Refer to Adjustment/Test, 27-41-00.)





25-061, 25-070 thru 25-341, except 25-337, modified for Horizontal Stabilizer Trim and Autopilot Improvement

Detail A

Aircraft 25-337, 25-342 and Subsequent

Detail B

Trim-in-Motion Detector Box Installation
Figure 201

8-96B

FLAPS - DESCRIPTION AND OPERATION

1. Description (See figure 1)
 A. The flap system is electrically controlled and hydraulically operated, provides lift to the wing when partially extended, and increases drag to reduce airspeed when fully extended.
 B. The flap system consists of two flaps, two flap actuators, a flap control valve, a restrictor, a pressure relief valve, a flap control valve switch, a system switch, a flap position indicator, and a flap position indicator calibration assembly. On Aircraft 25-181 and Subsequent, a check valve also is installed.
 C. Component Description
 (1) The single-slotted flaps are attached to the inboard rear wing spar by tracks, rollers, and hinges. An aural warning system is electrically connected to alert the pilot that the flaps have been fully extended without the landing gear being extended. Flap motion is a combination of outboard (aft) movement to increase wing area and tilting (down) movement to change the airflow section. Tracks are incorporated at WS 41 and 107 of each wing to direct flap travel. Maximum flap extension is 40° (+5°/-0°) down. Flaps will travel from maximum up to maximum down in 6 seconds or less. System operating pressure is 1500 (±50) psi.
 (2) Each flap actuator is attached to wing structure at WS 92 and to its respective flap sector. Access to the flap actuators is through access covers in the lower wing skin.
 (3) The flap control valve is a 4-way, 3-position, spring-loaded-to-center position, solenoid-operated valve which directs hydraulic pressure to each flap actuator to extend or retract the flaps. The valve operates on 28 vdc. The control valve is located on the top wing skin between wing spars 7 and 8. Access to the control valve is through the main landing gear wheel wells.
 (4) The restrictor is installed on the flap control valve in the pressure extend line. The restrictor restricts the hydraulic fluid flow, preventing rapid extension or retraction of the flaps.
 (5) The pressure relief valve is installed on the top wing skin adjacent to the flap control valve. The relief valve prevents damage to flap structure due to excessive airspeed with flaps down. The relief valve relieves pressure in excess of 1650 psi into the return line.
 (6) The flap control valve switch (LSW 29) is installed adjacent to the LH wing sector. The switch is actuated by the sector when the flaps are fully retracted. The switch automatically deenergizes the flap control valve when the flaps are fully retracted.
 (7) The Flap Switch, installed in the forward pedestal, is a three-position switch.

(8) The flap position indicator is installed in the center of the switch panel. The indicator is controlled by the flap position potentiometer (R18) in the LH wing. The indicator provides a visual indication of flap position for the crew.

(9) The calibration assembly is installed in the aft portion of the pedestal. The calibration assembly contains the circuits necessary to calibrate the flap position indicator. Small holes are provided in the pedestal so that adjustments can be made without removing the assembly.

(10) The check valve is installed on the flap control valve in the incoming hydraulic pressure line. The check valve prevents flap blow-up, on approach, should a plumbing failure occur in the pressure line between the flap control valve and the pressure source.

2. Operation (See figures 2 and 3)

A. Aircraft 25-070 and Subsequent.

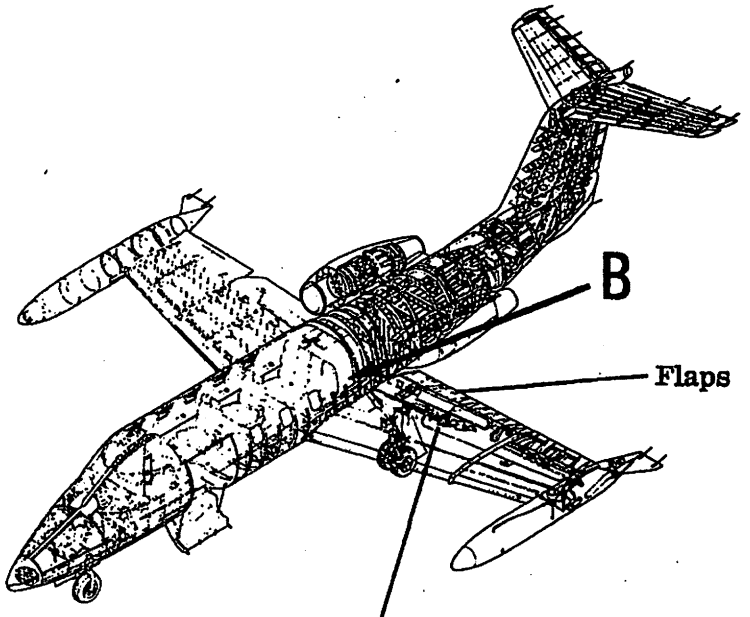
NOTE: On extended parking time with power shut off, the flaps may droop and spoilers rise. On starting either engine or the auxiliary pump, the flaps and spoilers will return to their positions prior to power shutoff provided their control switch settings have not been changed.

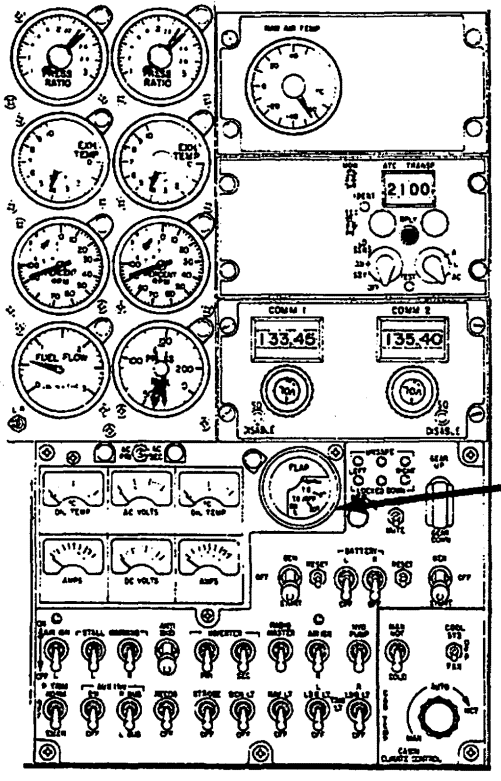
(1) Setting the Flap Switch to FLAPS DN energizes a solenoid which positions the flap control valve to direct hydraulic fluid to the downside of the flap actuators. This in turn actuates the system to begin flap extension.

(2) When the Flap Switch is set to the off (center) position, the solenoid is deenergized and the flap control valve returns to neutral. This cuts off all hydraulic flow to the actuator and stops the flaps at the desired position.

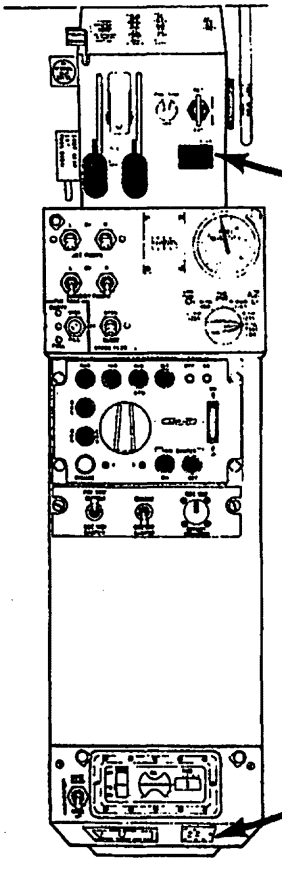
(3) Setting the Flap Switch to FLAPS UP energizes the solenoid and positions the flap control valve selector to direct hydraulic fluid to the up side of the flap actuators. This in turn actuates the system to begin flap retraction. When the flaps are fully retracted, the flap control valve switch is actuated, opening the electrical circuit to deenergize the solenoid and allows the flap control valve to return to neutral.

(4) The aural warning horn will sound when the flaps are extended more than 25° with the gear retracted. The aural warning horn will sound until the gear is down and locked or the flaps are retracted above 25°.



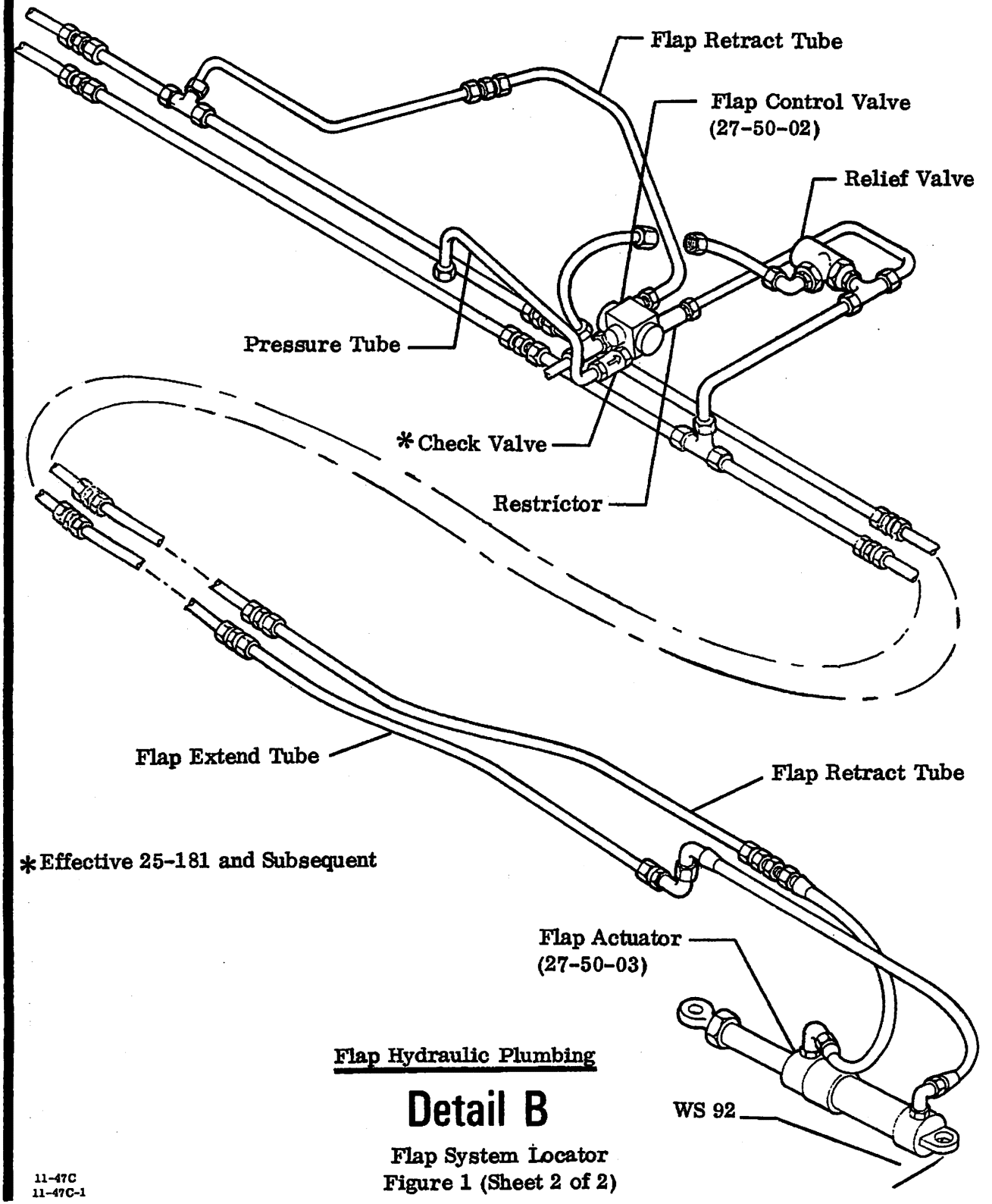


Center Instrument Panel



Detail A

Flap System Locator
Figure 1 (Sheet 1 of 2)



*Effective 25-181 and Subsequent

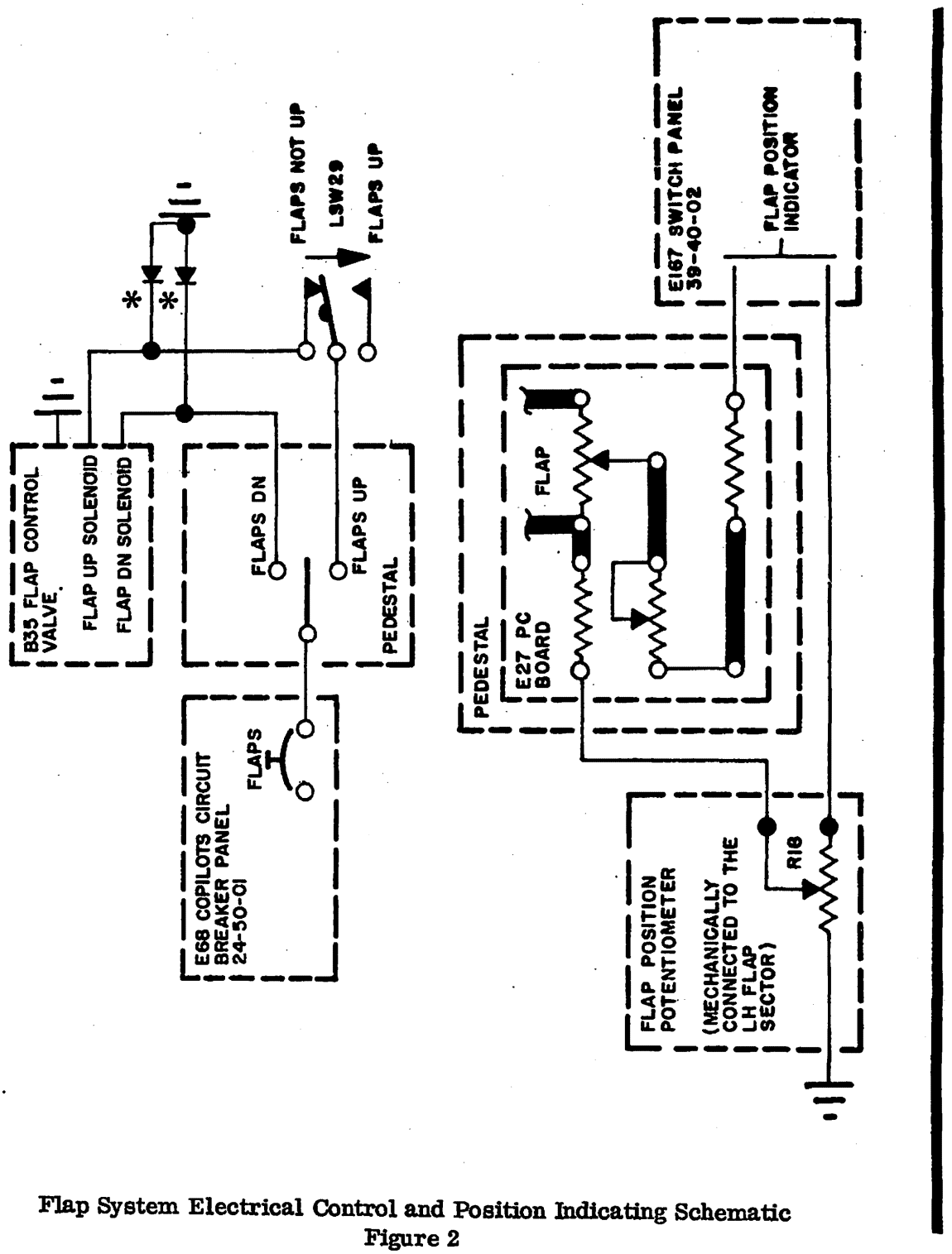
Flap Hydraulic Plumbing

Detail B

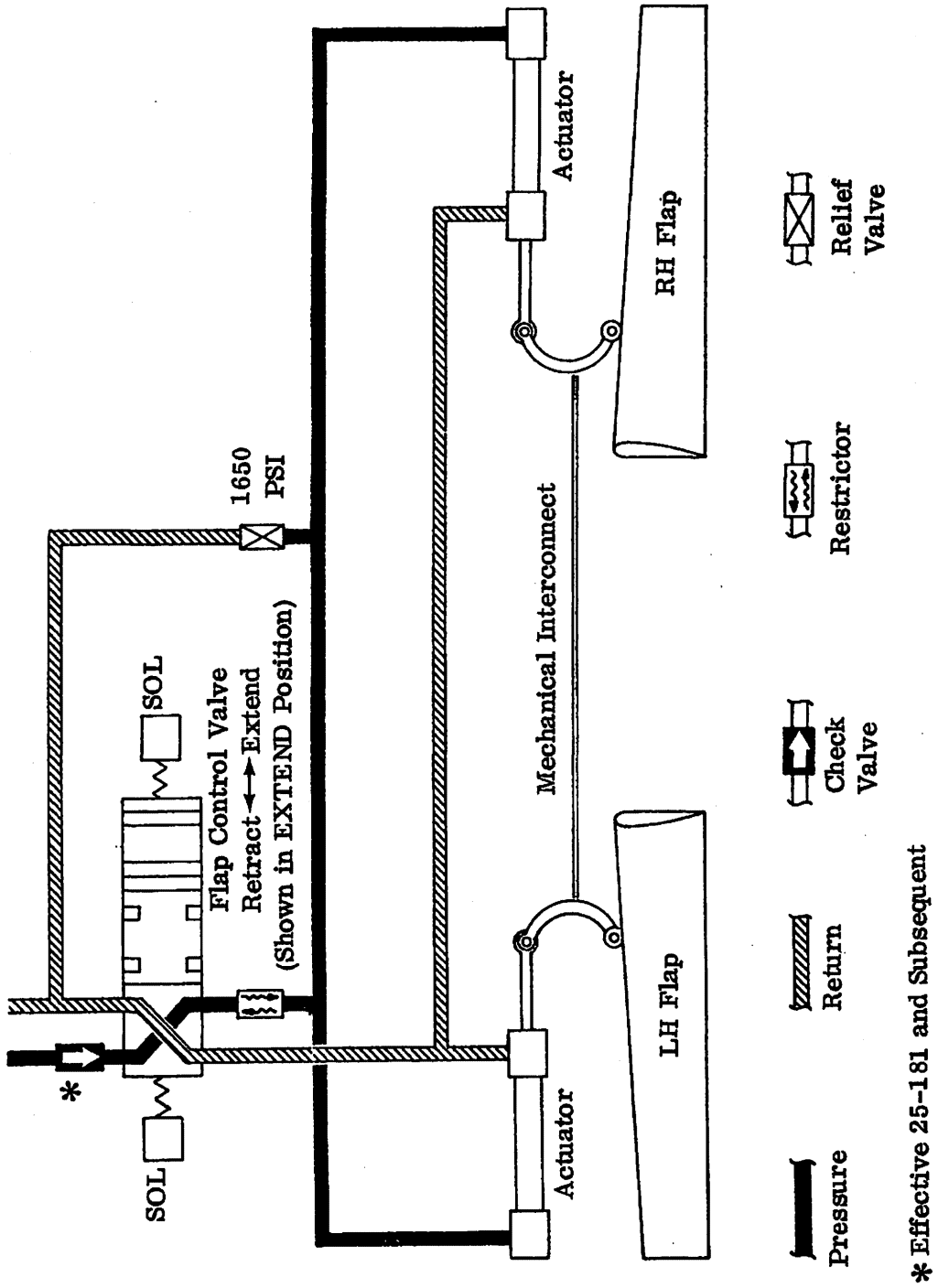
Flap System Locator
Figure 1 (Sheet 2 of 2)

11-47C
11-47C-1

* Diodes added effective Aircraft 25-173 and Subsequent



Flap System Electrical Control and Position Indicating Schematic
Figure 2



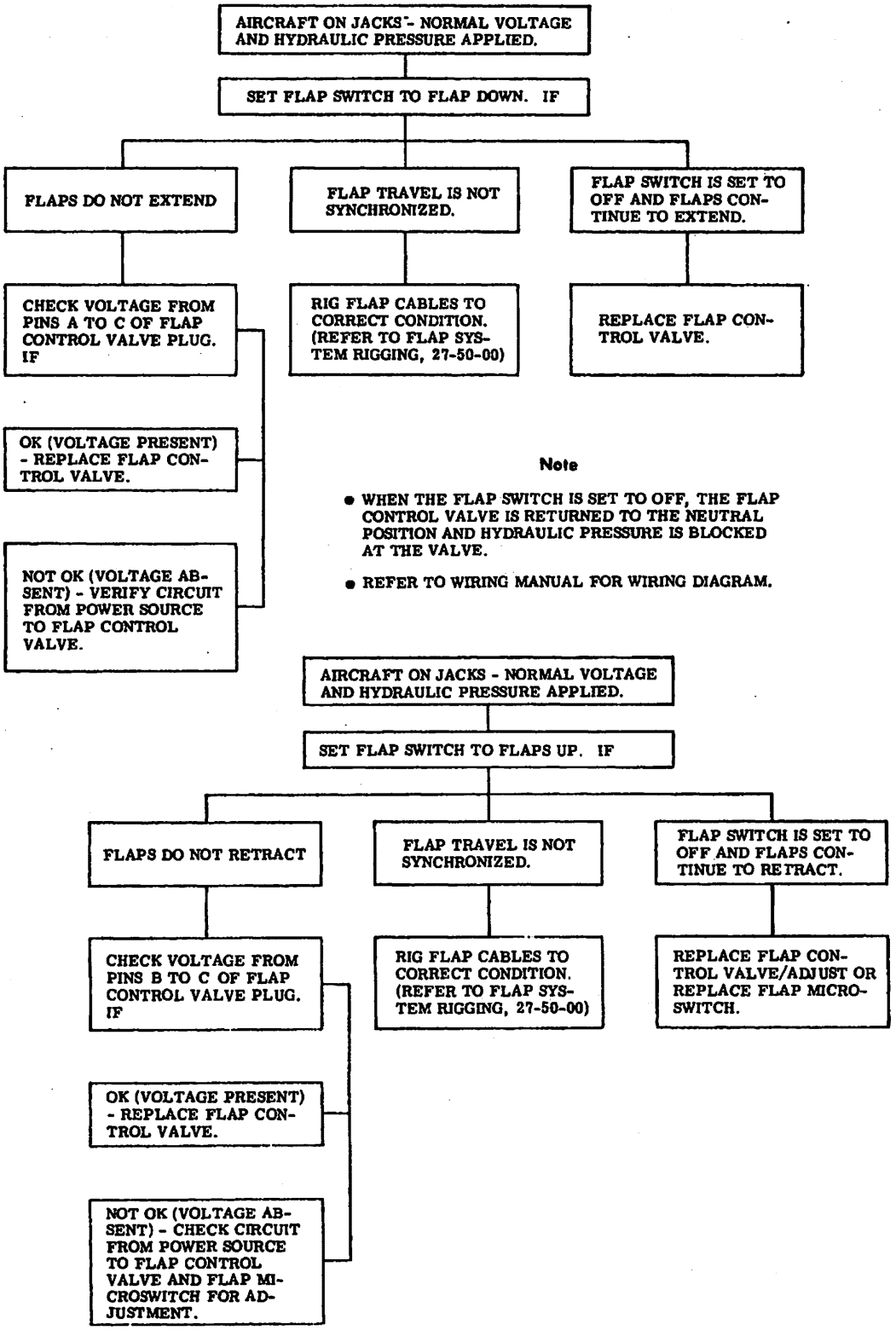
Flap Hydraulic System Control Schematic
Figure 3

FLAPS - MAINTENANCE PRACTICES

1. General
 - A. Maintenance practices on the flap system consists of removal and replacement of defective components.
 - B. When installing a new flap it is necessary to trim the flap fairing to proper fit. Two installation procedures are provided, one for installing the existing flap and one for a new flap.

2. Trouble Shooting
 - A. Figure 201 is provided as an aid in isolating trouble in the flap system.

3. Removal/Installation
 - A. Remove Flap (See figure 202)
 - (1) Extend flaps and spoilers.
 - (2) Remove access cover from underside of wing and insert rigging pin in flap sector.
 - (3) Disconnect flap center bonding jumper from wing aft spar. On Aircraft 25-294 and Subsequent, disconnect inboard and outboard bonding jumpers from bottom of upper wing skin.
 - (4) Disconnect flap push-pull tube from sector.
 - (5) Remove bolts securing flap to flap nose track releasing rollers, spacers, and washers from flap nose track.
 - (6) Check all track rollers for scoring and excessive wear. Replace as necessary.
 - (7) Check all flap tracks for rust.
 - (8) Check push-pull tube end bearings. Replace as necessary.
 - (9) Check bearings in flap tracks. Replace as necessary.
 - (10) If flap is to be replaced remove flap tracks, push-pull tube and disconnect bonding jumper from existing flap.
 - B. Install Existing Flap (See figure 202)
 - (1) Prior to installation of flap check all components for excessive wear.
 - (2) Clean flap tracks and flap nose tracks.
 - (3) Lubricate flap track rollers with MIL-G-81322 grease using a zerk gun and an Alemite 90° rubber tip adapter (P/N 327265).
 - (4) Align flap and flap tracks with flap track rollers and push-pull tube with push-pull tube access hole in wing rear spar.
 - (5) Insert flap tracks on rollers and push-pull tube into access hole assuring flap tracks align on forward rollers and flap nose track is between flap nose hinge brackets.
 - (6) With flap nose tracks on flap nose hinge brackets, install nose flap track rollers, spacers, and washers using attaching parts.
 - (7) Connect flap center bonding jumper at rear wing spar and push-pull tube to flap sector. On Aircraft 25-294 and Subsequent, connect inboard and outboard bonding jumpers to bottom of upper wing skin.



Flap System Trouble Shooting
Figure 201

(8) Attach flap push-pull tube to sector by inserting bolt from the bottom side and secure with nut, washer and cotter pin.
(9) Rig flaps. Refer to 27-50-00, Adjustment/Test.
(10) Remove rigging pin from sector and install access cover.

C. Install Replacement Flap (See figure 202)
(1) Prior to installation of flap check all components for excessive wear.

NOTE: Attaching hardware not supplied with a new flap should be removed from the old flap and used on the new flap.

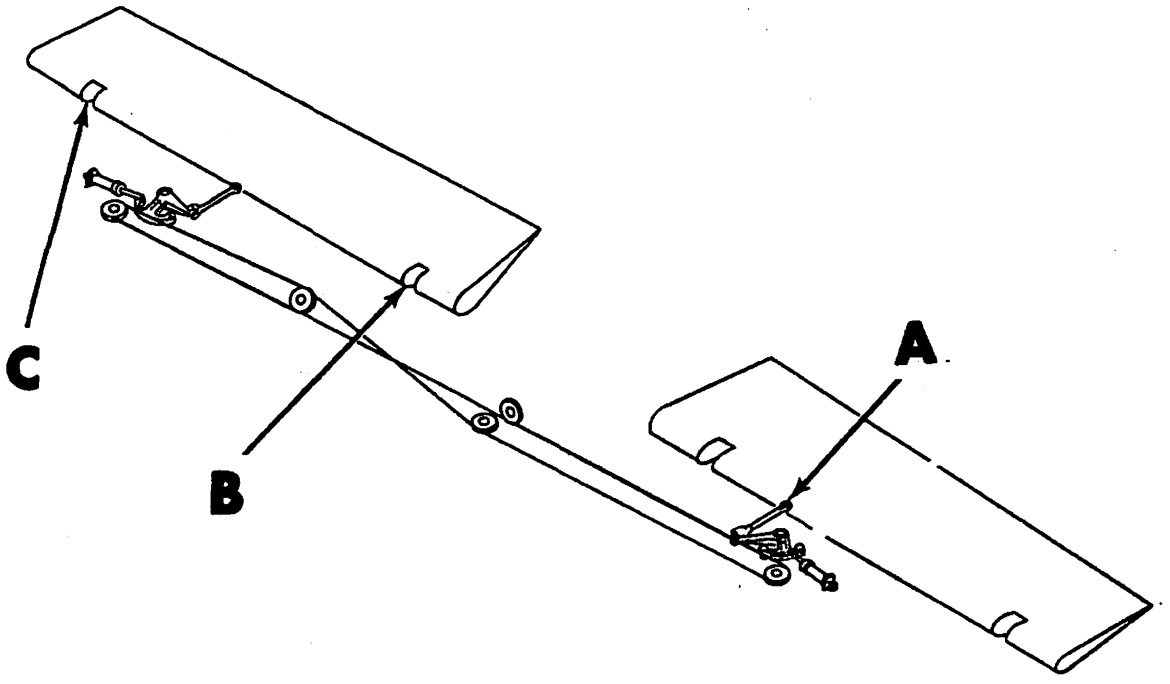
(2) Clean flap tracks and flap nose tracks.
(3) Lubricate flap track rollers with MIL-G-81322 grease using zerk gun with Alemite 90° rubber tip adapter (P/N 327265).
(4) Install flap tracks on flaps using attaching parts removed from old flap.
(5) Connect bonding jumper and push-pull tube to flap.
(6) Align flap and flap tracks with flap track rollers and push-pull tube with push-pull tube access hole in wing rear spar.
(7) Insert flap tracks on rollers and push-pull tube into access hole assuring flap tracks align on forward rollers and flap nose track is between flap nose hinge brackets.
(8) Raise flap by hand to the retracted position.

NOTE: Flap cannot be raised to the full retract position until after the flap fairing has been trimmed.

(9) Measure and mark flap fairing so that it may be trimmed to the gap shown in figure 202.
(10) Lower flap and trim along the trim line previously established.
(11) Raise flap by hand to the full retract position and check for proper gap between fuselage fairing and flap fairing. Trim as necessary to obtain uniform gap.
(12) With flap nose tracks on flap nose hinge brackets, install nose flap track rollers, spacers, and washers using attaching parts.
(13) Connect flap bonding jumper to rear wing spar.
(14) Attach flap push-pull tube to sector by inserting bolt from the bottom side and secure with nut, washer and cotter pin.
(15) Rig flaps. Refer to 27-50-00, Adjustment/Test.

4. Adjustment/Test
A. Flap system rigging consists of synchronizing the flaps, adjusting cable tension and adjusting the flap limit switch.
B. Tools and Equipment

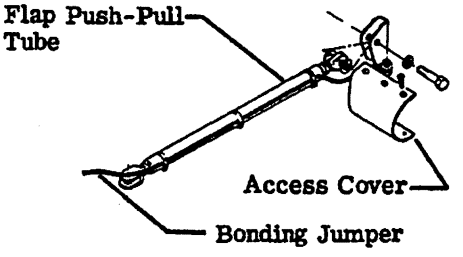
NAME	NUMBER	MANUFACTURER	USE
Tensiometer	2471007	Gates Learjet Corp.	Adjust Cable Tension



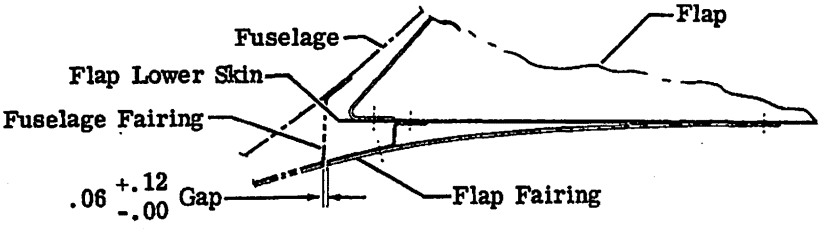
CAUTION

1. Be sure heads of these bolts are located as shown. If heads of bolts are NOT on side shown, bolts will not clear flaps when in the retracted position.
2. DO NOT lubricate the Flap Nose Rollers.

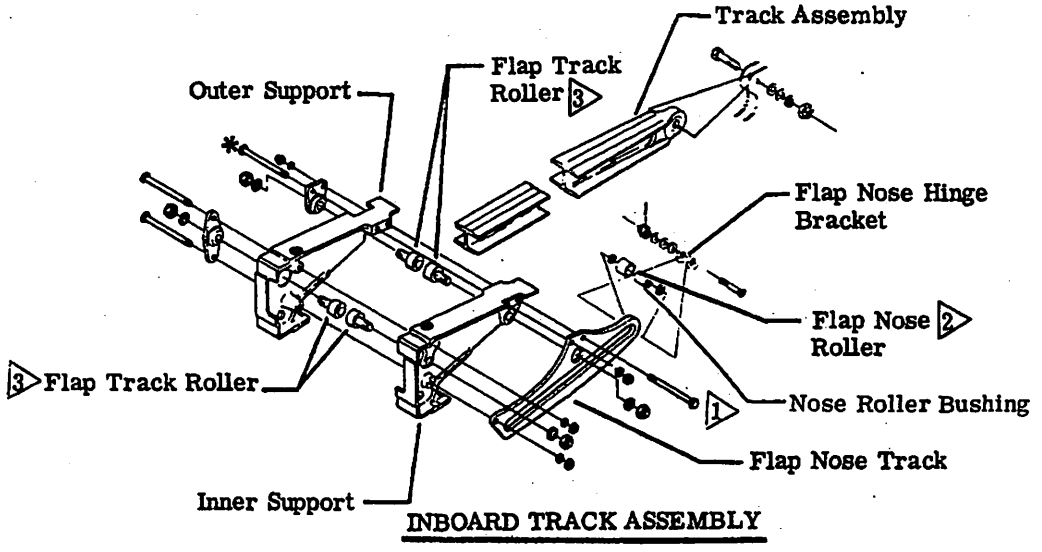
3. Lubricate Flap Track Rollers (Refer to 12-21-02)



Detail A

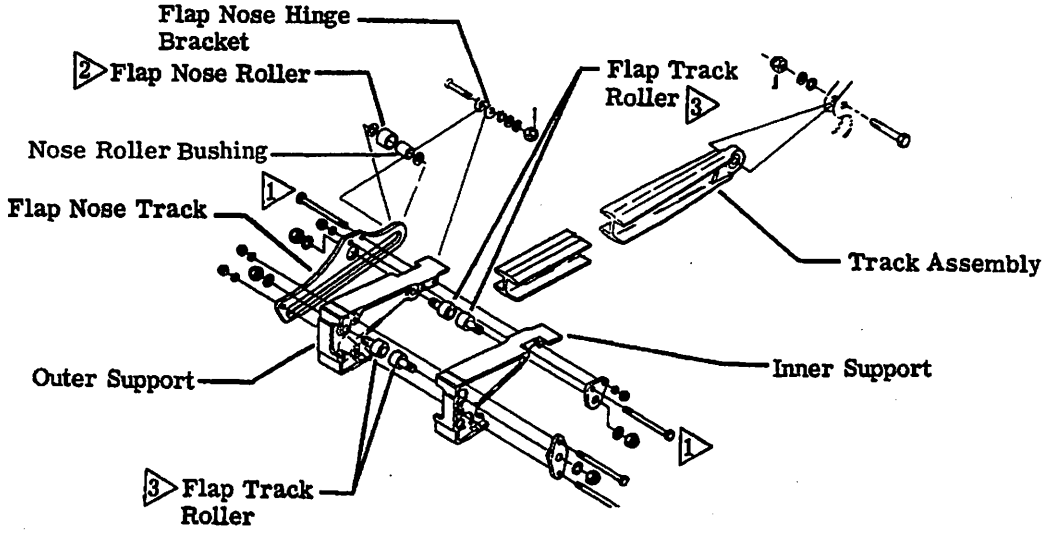


Flap Installation
Figure 202 (Sheet 1 of 2)



INBOARD TRACK ASSEMBLY

Detail B



OUTBOARD TRACK ASSEMBLY

Detail C

Flap Installation
Figure 202 (Sheet 2 of 2)

C. Rig Flap System (See figure 203)

(1) With flaps in retract position, remove flap sector access cover, align rigging pin holes in sector with holes in bracket, and insert rigging pin.

(2) Disconnect actuator rod ends from both left and right sectors.

(3) Check for 0.03 inch minimum clearance between sectors and wing spar 8.

(4) Adjust flap push-pull tubes until flap rollers contact forward end of flap tracks.

(5) Adjust cable tension to 100 (±25) pounds at 75°F. (Refer to 27-00-00 for cable tension temperature-load correction.)

(6) With hydraulic actuators fully retracted (bottomed internally), adjust rod end terminals and attach to sectors.

(7) Remove bolts attaching actuator rod end terminals to sectors and turn rod ends one-half turn clockwise (shortening retracted length of cylinders approximately 0.021 inch).

(8) Attach rod end terminals to sectors and adjust flap control valve microswitch to actuate at exact position determined by sectors with rigging pins installed.

(9) Safety wire flap cable turnbuckles and remove rigging pins.

(10) Using Flap Switch on center pedestal, check for proper travel. Full down travel should be 40° (+5°/-0°).

(11) With flaps down, spanwise movement (end play) shall be a maximum of 0.34 inch.

(12) Install access covers.

(13) Perform operational check of stall warning system (27-31-00), spoiler warning system (27-60-00), and landing gear aural warning system (Chapter 32). Make adjustments as required.

5. Inspection/Check

A. Operational Check of Flap System

(1) Lower tailcone access door.

(2) Connect a hydraulic power unit to pressure and return quick-disconnects.

(3) Set Battery Switch(es) to ON.

(4) Set Flap Switch to FLAP DN.

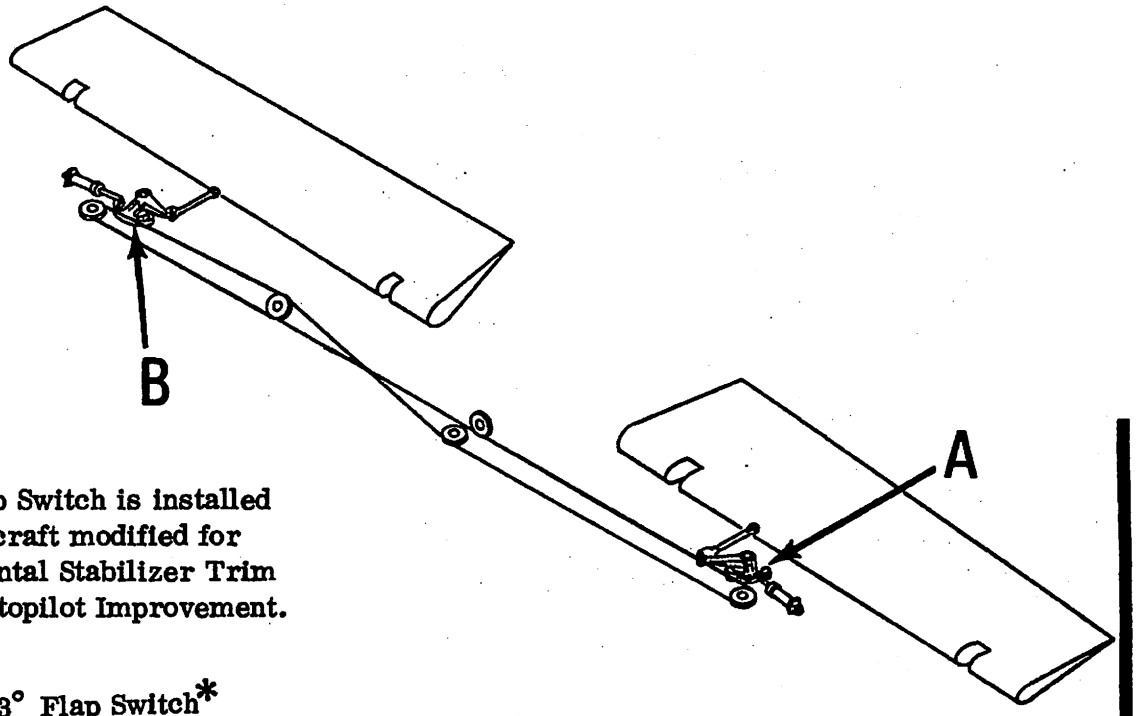
(5) During flap extension, observe flap position indicator. Stop flaps at each marked interval until flaps are full down.

(6) Set Flap Switch to FLAP UP.

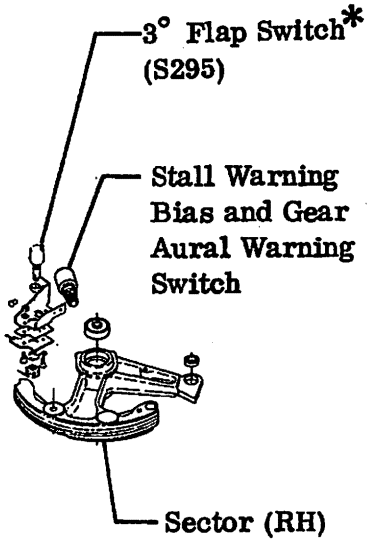
(7) During flap retraction, observe that indicator pointer movement from DN to UP is smooth and uninterrupted. Set Battery Switch(es) to OFF.

(8) Disconnect hydraulic power unit and install caps on hydraulic pressure and return quick-disconnects.

(9) Secure tailcone access door.

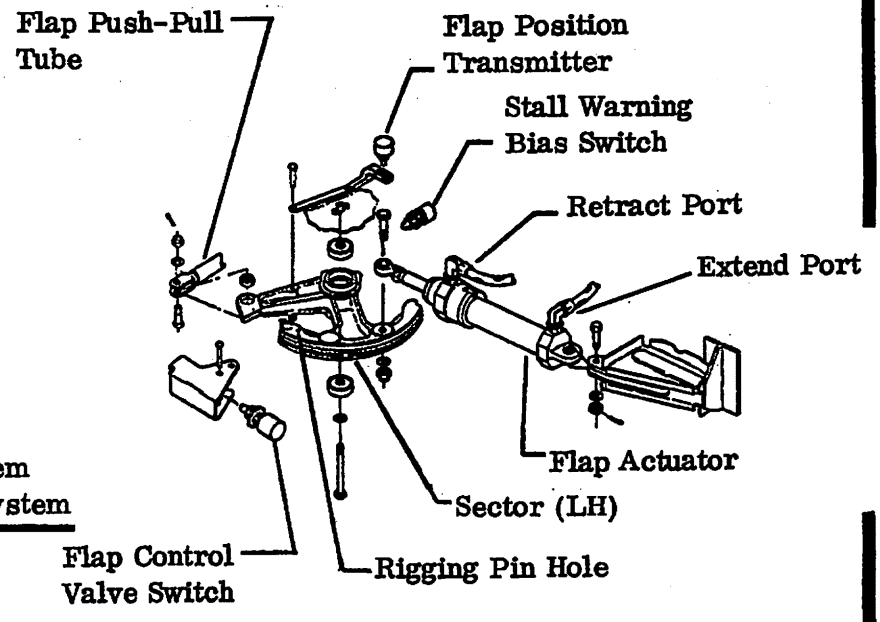


*3° Flap Switch is installed on Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement.



Aircraft with 20° Flap System and Aircraft with 8° Flap System

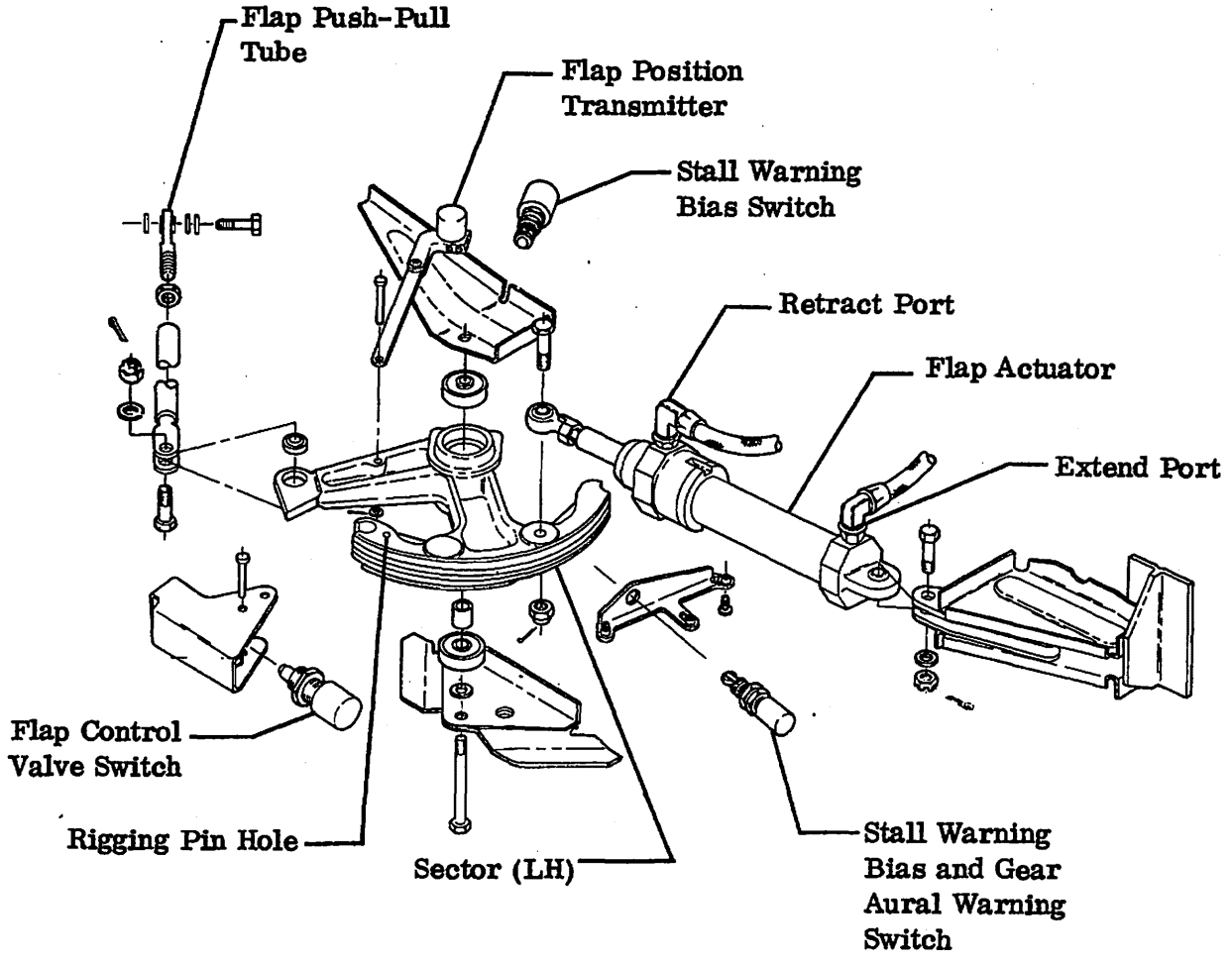
Detail B



Aircraft with 20° Flap System Only

Detail A

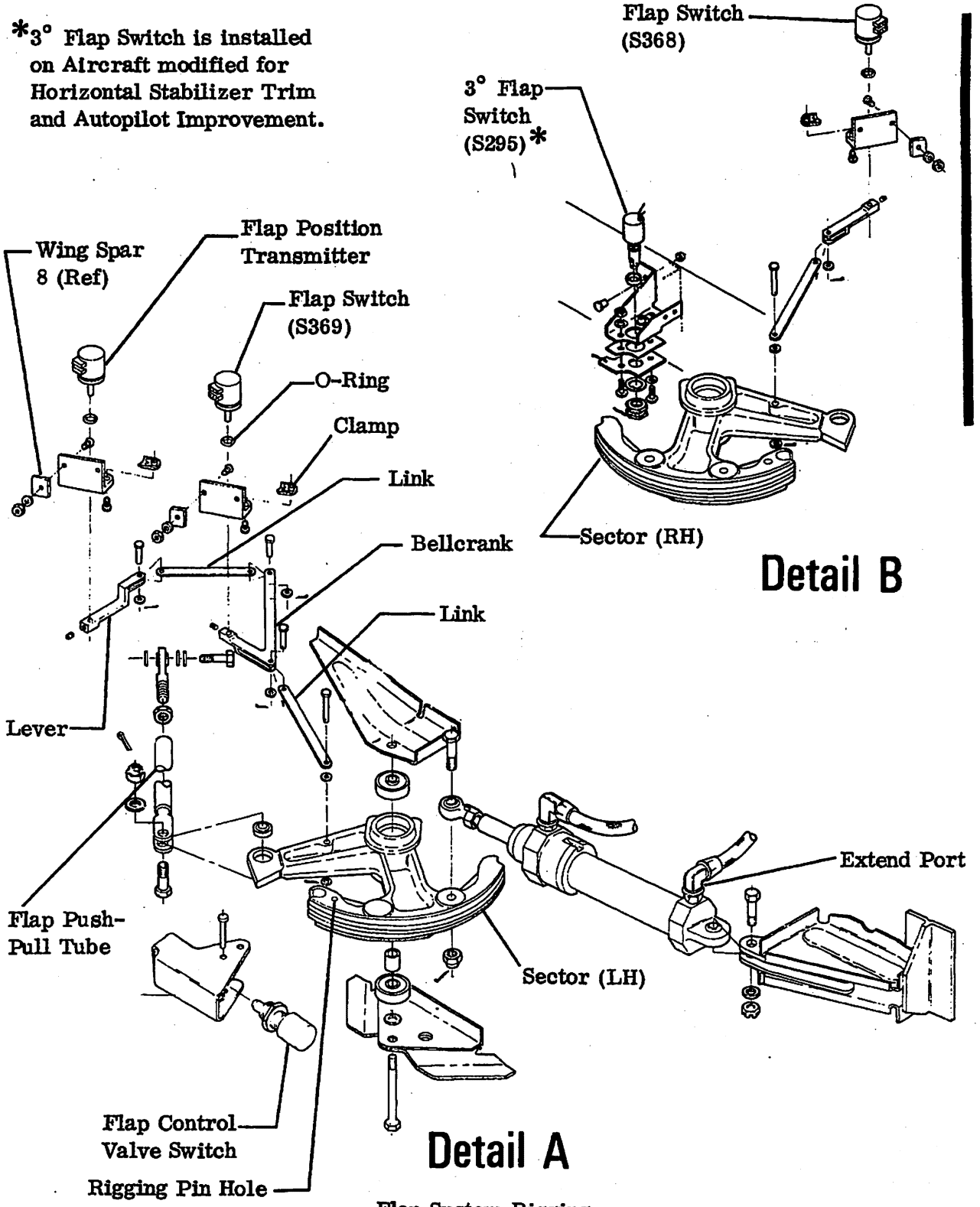
Flap System Rigging
Figure 203 (Sheet 1 of 3)



Detail A

Flap System Rigging
Figure 203 (Sheet 2 of 3)

*3° Flap Switch is installed on Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement.



8-6C-2
8-79B

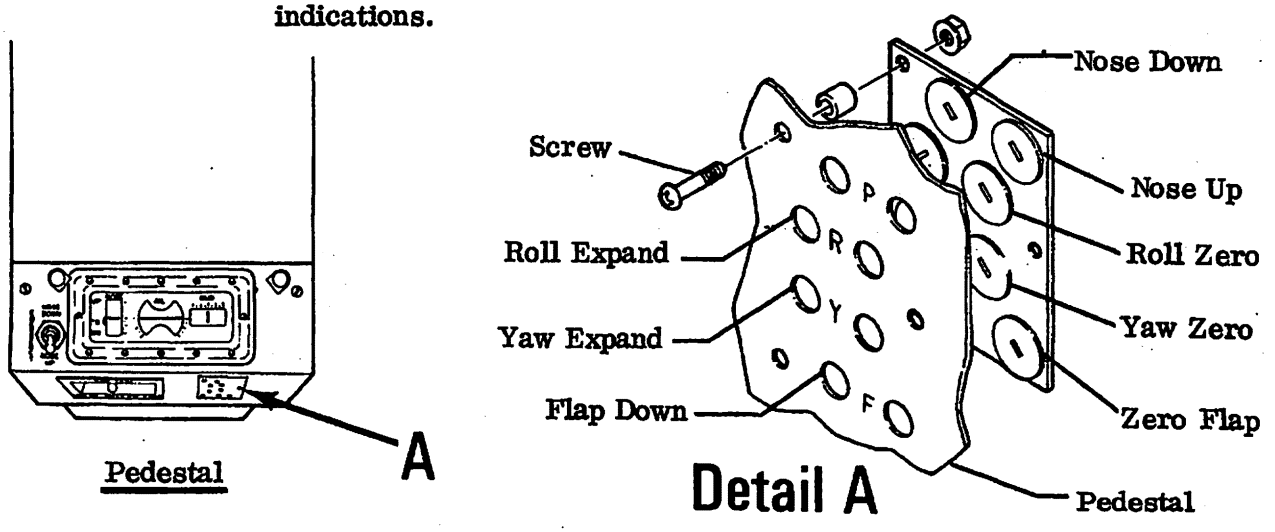
Flap System Rigging
(Sheet 3 of 3)

EFFECTIVITY: Aircraft with Reduced Approach Speed System

FLAP POSITION INDICATOR CALIBRATION ASSEMBLY - MAINTENANCE PRACTICES

1. Removal/Installation
 - A. Remove Flap Position Indicator Calibration Assembly (See figure 201)
 - (1) Remove electrical power from aircraft.
 - (2) Remove equipment as required from lower aft end of pedestal to gain access to calibration assembly.
 - (3) Disconnect electrical wiring from calibration assembly.
 - (4) Remove attaching parts and calibration assembly from pedestal.
 - B. Install Flap Position Indicator Calibration Assembly (See figure 201)
 - (1) Install calibration assembly and secure with attaching parts.
 - (2) Connect electrical wiring to calibration assembly.
 - (3) Install and secure previously removed equipment.
 - (4) Restore electrical power to aircraft.
 - (5) Calibrate the flap position indicator.

2. Adjustment/Test, (See figure 201)
 - A. Calibrate flap position indicator as follows:
 - (1) Insert small screwdriver through holes in center pedestal to adjust potentiometers.
 - (2) Actuate flaps to full up position.
 - (3) Adjust Zero Flap potentiometer to obtain flaps UP indication on flap position indicator.
 - (4) Actuate flaps to full down position.
 - (5) Adjust Flap Down potentiometer to obtain flaps DN indication on flap position indicator.
 - (6) Cycle flaps while checking respective indication on flap position indicator.
 - (7) Readjust Flap Down potentiometer as necessary to obtain coordinated indications.



Position Indicator Calibration Assembly
Figure 201

8-107A

FLAP CONTROL VALVE - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Flap Control Valve (See figure 201)

(1) Depressurize hydraulic system.

(2) Remove electrical power from aircraft.

(3) Lower inboard landing gear door. Remove safety wire and disconnect electrical connector from valve.

NOTE: On Aircraft 25-181 and Subsequent, a check valve is installed in place of the elbow.

(4) Disconnect pressure tube from elbow or check valve at pressure port of valve.

(5) Disconnect flap valve return tube and brake return line from tee at return port of valve.

(6) Disconnect flap retract tube from union on valve.

(7) Disconnect tube from restrictor.

(8) Remove attaching parts and valve from mounting bracket and remove valve from aircraft.

(9) Remove elbow or check valve, tee, restrictor, and union from ports of valve.

B. Install Flap Control Valve (See figure 201)

(1) Inspect elbow or check valve, tee, restrictor, and union removed from valve. If serviceable, install on new valve using new O-ring and packings.

(2) Position valve on mounting bracket and check that elbow or check valve and tee are clocked to allow connection of hydraulic lines.

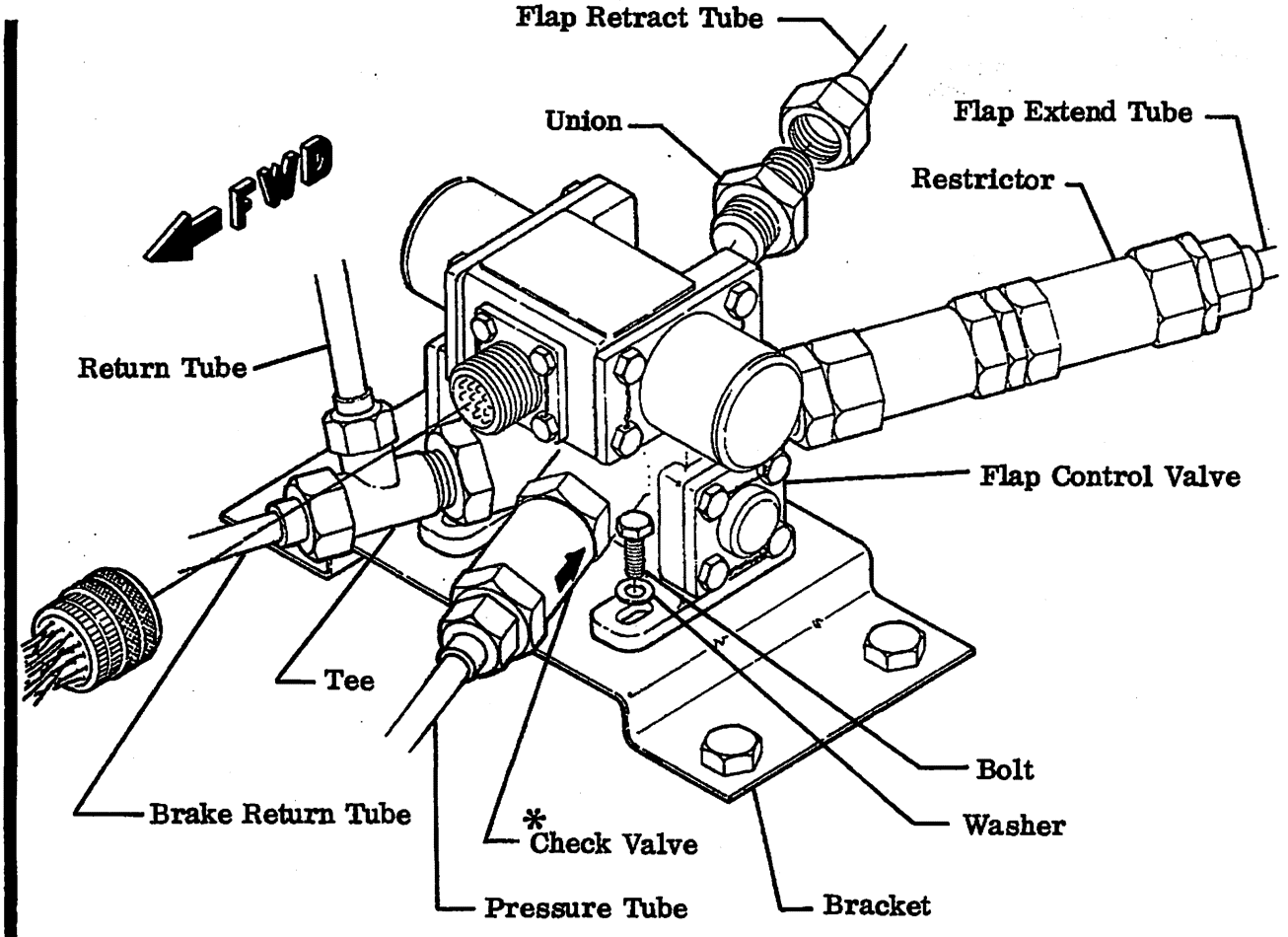
(3) Install valve to mounting bracket and secure with attaching parts.

(4) Connect hydraulic tubes to ports from which they were removed.

(5) Connect electrical connector to valve and safety wire.

(6) Restore electrical power to aircraft.

(7) Pressurize hydraulic system. Check for leaks and flap operation.



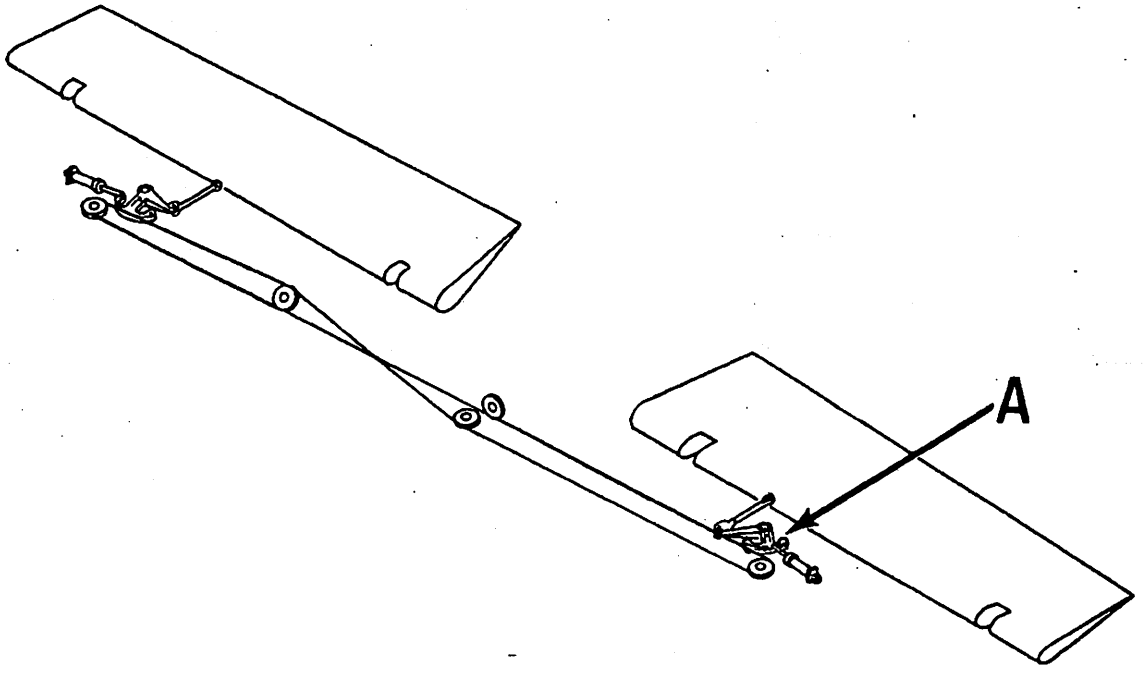
*Effective 25-181 and Subsequent.
On Prior Aircraft, an elbow is installed.

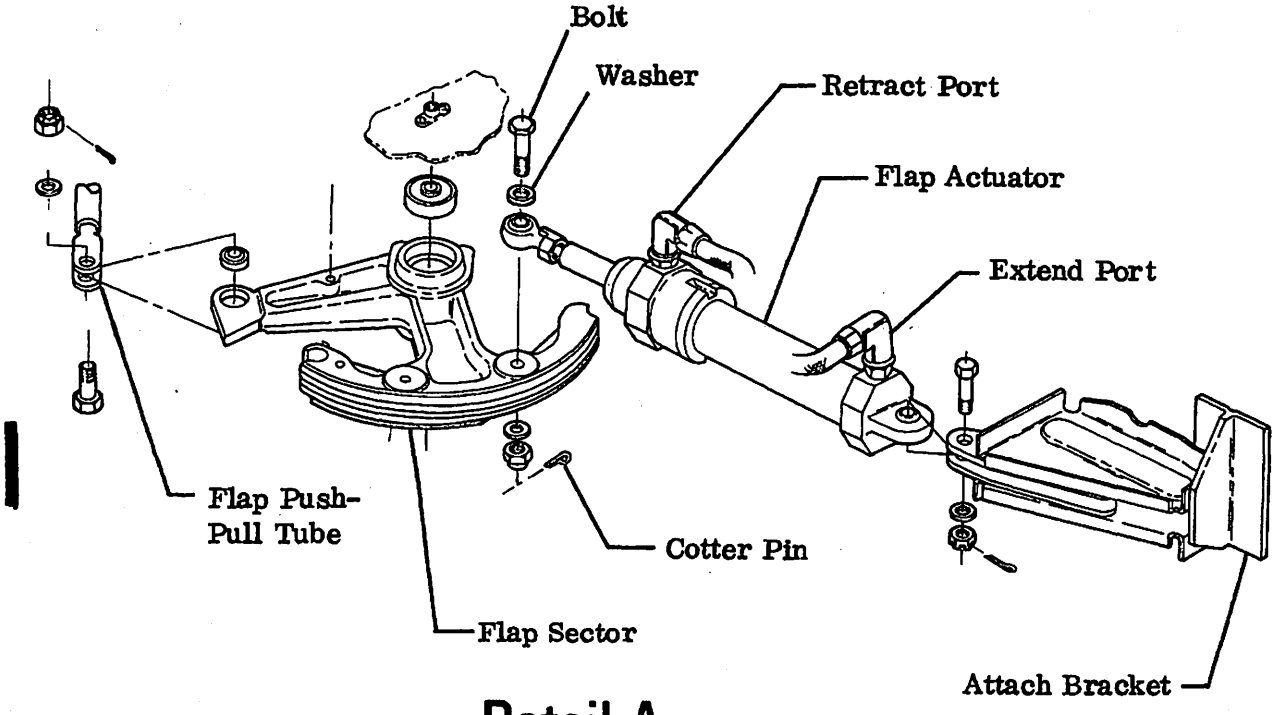
Flap Control Valve Installation
Figure 201

FLAP ACTUATOR - MAINTENANCE PRACTICES

1. Removal/Installation
 - A. Remove Flap Actuator (See figure 201)
 - (1) Depressurize the hydraulic system. Refer to Chapter 29.
 - (2) Remove flap actuator access cover.
 - (3) Disconnect and cap hydraulic lines at actuator.
 - (4) Disconnect actuator from attach bracket and sector and remove actuator from aircraft.
 - B. Install Flap Actuator (See figure 201)
 - (1) Position flap actuator between attach bracket and sector and secure with attaching parts.
 - (2) Remove caps and connect hydraulic lines to actuator.
 - (3) Rig flap system. Refer to 27-50-00, Adjustment/Test.
 - (4) Bleed hydraulic system. Refer to Chapter 29.
 - (5) Install access cover.

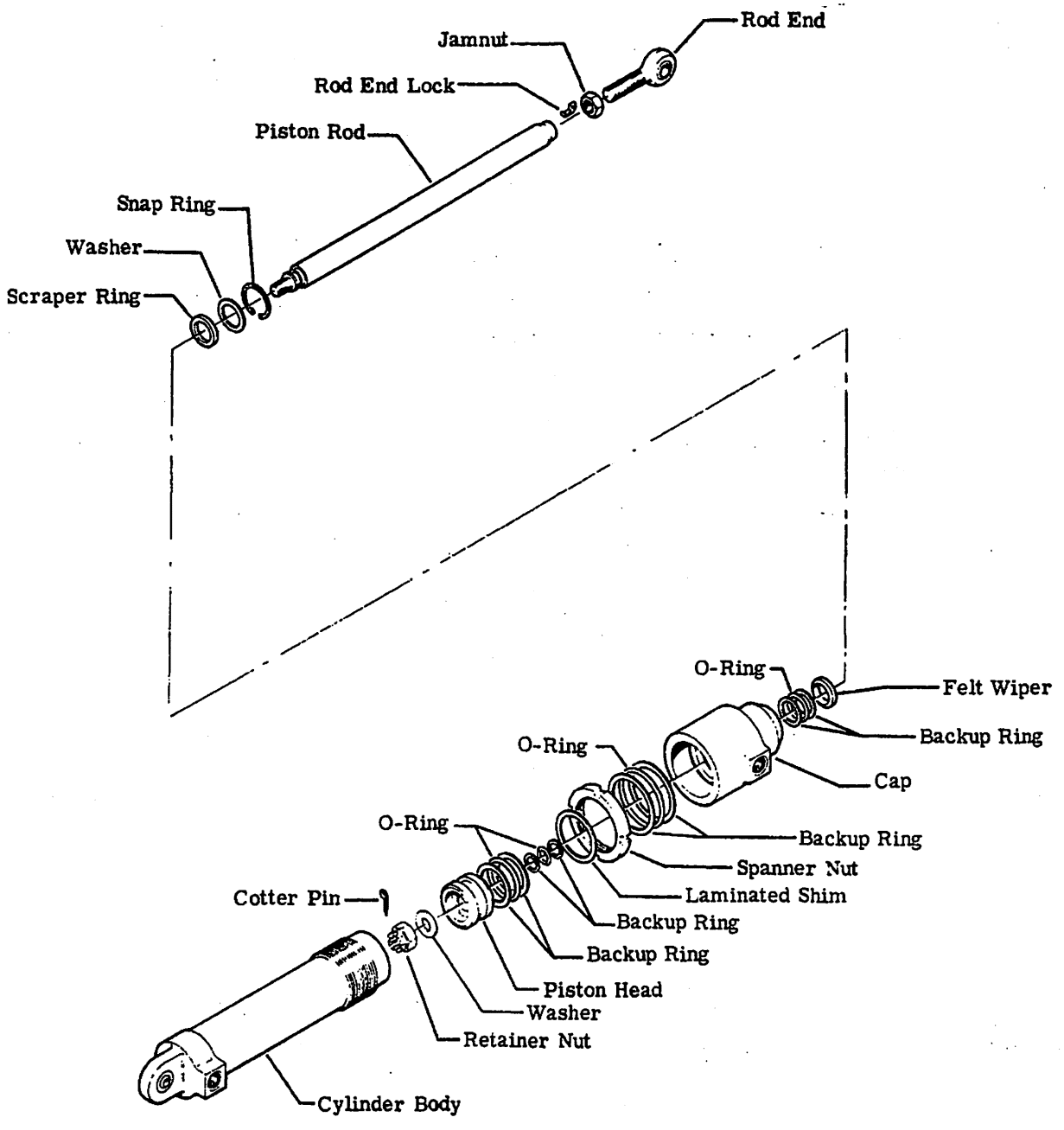
2. Approved Repairs
 - A. The following procedure lists disassembly and assembly of flap actuator necessary to replace O-rings, backup rings and felt wiper. New O-rings, backup rings and felt wiper must be installed at each assembly. These new items are to be immersed in MIL-H-5606 hydraulic fluid immediately prior to installation.
 - B. All O-ring grooves are to be free of gouges, scratches, and burrs. Assure all items are thoroughly clean prior to installation.
 - C. Disassembly
 - (1) Disassemble Flap Actuator (See figure 202)
 - (a) Cut safety wire and loosen jamnut sufficiently to release rod end lock. Remove rod end from piston rod.
 - (b) Cut safety wire and loosen spanner nut. Remove cap from cylinder body. Remove piston rod from cylinder body.
 - (c) Remove cotter pin and retainer nut securing piston head to piston rod.
 - (d) Remove snap ring from cap. Remove all backup rings, O-rings, washer, laminated shim, scraper ring, and felt wiper from piston rod, piston head, cylinder body and cap.
 - (e) Clean all parts in Stoddard solvent or equivalent. Dry parts with clean, dry compressed air.





Detail A

Flap Actuator Installation
Figure 201



Flap Actuator Assembly
Figure 202

(f) Inspect cylinder body, cap, piston rod, and piston head for nicks, scratches, or excessive wear. Buff out minor defects with an oil stone, then polish with crocus cloth.

D. Assembly

(1) Assembly Flap Actuator (See figure 202)

(a) Immerse all new items in MIL-H-5606 hydraulic fluid.

(b) Assemble O-ring and backup rings on piston rod.

(c) Install piston head on piston rod and secure with retainer nut and cotter pin. Assemble O-ring and backup rings on piston head.

(d) Apply a thin coat of MIL-H-5606 hydraulic fluid to piston head and insert piston rod with piston head attached into cylinder body.

(e) Assemble O-rings, backup rings, laminated shim, and felt wiper in cap. Screw cap on cylinder body as far as possible, assuring that the extend and retract ports are in line.

(f) Tighten spanner nut against cap and safety wire.

(g) Assemble scraper ring, washer, and snap ring in cap. Install rod end with jamnut and rod end lock in piston rod.

(h) Connect actuator to a hydraulic test bench. Pressure check actuator, both extend and retract positions, to 2250 psi for 3 minutes. Leakage should not exceed one drop in the 3-minute period.

(i) With the actuator retracted, adjust for 12.950 (±0.010) inches between centerline of actuator bearing and rod end. Safety wire rod end lock.

FLAP POSITION INDICATOR - MAINTENANCE PRACTICES

1. Removal/Installation

A. Remove Flap Position Indicator (See figure 201)

(1) Remove electrical power from aircraft.

(2) Remove switch panel sufficiently to gain access to position indicator. (Refer to Chapter 39.)

(3) Disconnect wiring from position indicator.

(4) Loosen and remove attaching parts and position indicator from switch panel.

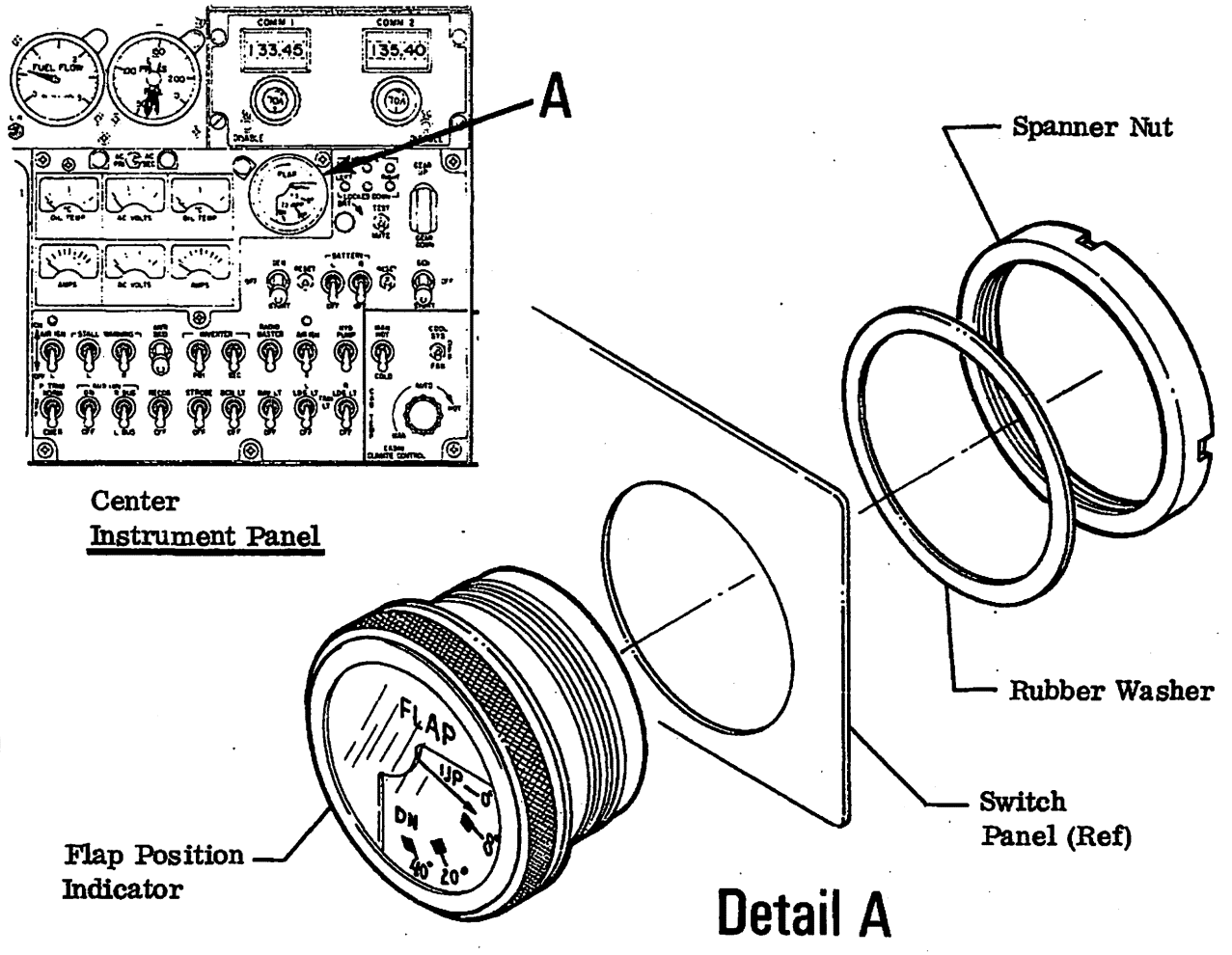
B. Install Flap Position Indicator (See figure 201)

(1) Install position indicator in switch panel and secure with attaching parts.

(2) Connect electrical wiring to position indicator.

(3) Install and secure switch panel. (Refer to Chapter 39.)

(4) Calibrate position indicator. (Refer to 27-50-01.)



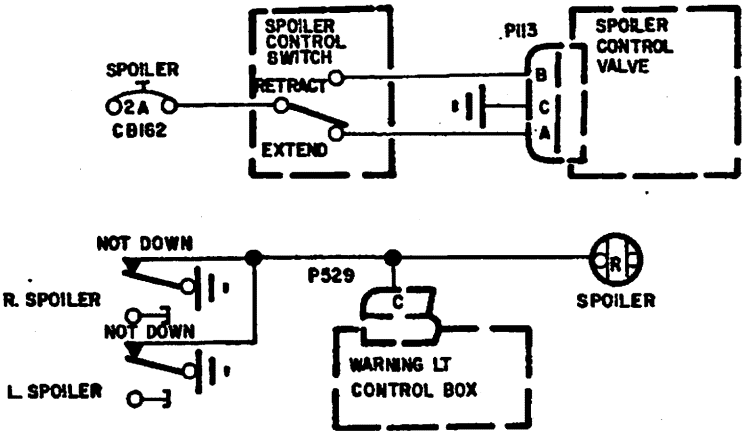
Flap Position Indicator Installation
Figure 201

SPOILER - DESCRIPTION AND OPERATION

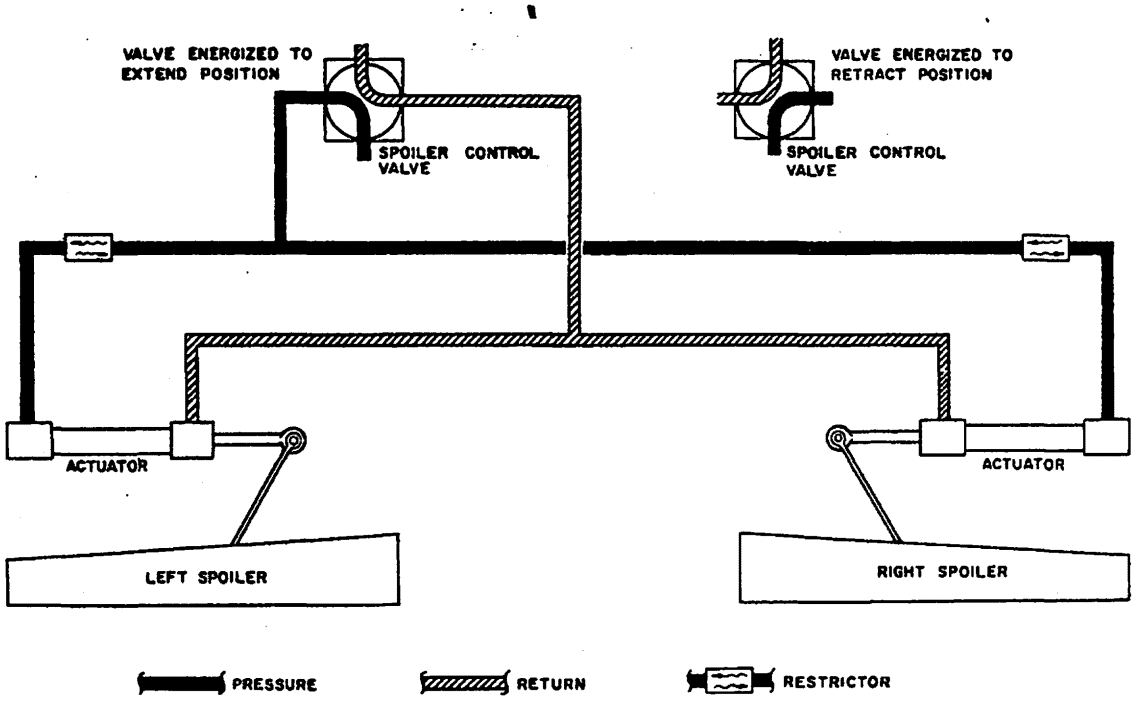
1. Description
 A. The spoiler system is electrically controlled and hydraulically operated. The system consists of a spoiler switch on the cockpit center pedestal, a solenoid-operated spoiler control valve, two restrictors, two actuators, and a (red) spoiler warning light on the glareshield warning light panel.
 B. The spoilers are located on the upper surface of the wing forward of the flaps and, when hydraulically actuated, hinge up at an angle of 40° (-0°/+7°). Full extension is 47°. However, as air speed increases, a pressure relief allows the spoilers to "blow down" to a lesser extension angle. The spoiler travel arc is governed by the extension of the actuator rod end; this effort being transferred to the spoiler by the actuator arm assembly. The spoilers are controlled by a switch on the cockpit center pedestal.
 C. Each spoiler is extended and retracted by a hydraulic actuator attached to the spoiler control arm and the wing structure. The actuator incorporates an internal locking arrangement consisting of steel locking balls and a spring-loaded plunger. Setting the Spoiler Switch to EXT directs hydraulic pressure to the actuator. Hydraulic pressure of approximately 150 psig for 6600135-1 actuator (225 psig for 6600135-2) then overrides the spring-loaded plunger, releasing the steel locking balls, to allow the actuator to extend the spoilers. Setting the Spoiler Switch to RET drives the piston containing the steel locking balls into the actuator head end. The steel balls engage a lip in the head end and are retained in the locked position by the spring-loaded plunger.

2. Operation
 A. Positioning the control switch to either EXT or RET energizes an electrically actuated hydraulic control valve which directs hydraulic pressure to both spoiler actuators. Adjustable switches, located adjacent to each spoiler actuator, light the SPOILER warning light when the spoilers are not down and locked.

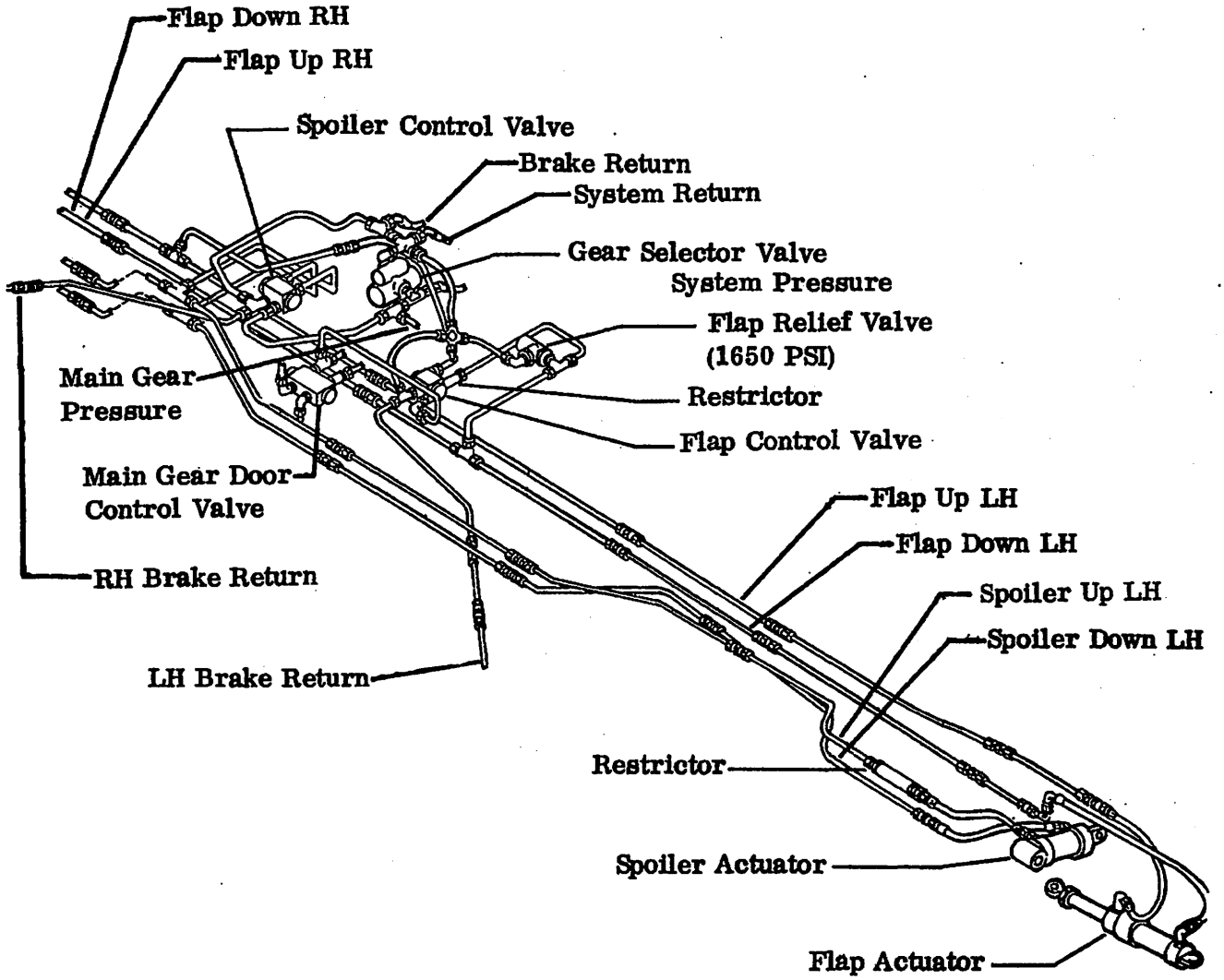
 NOTE: On extended parking time with power shut off, the flaps may droop and spoilers rise. On starting either the engines or the auxiliary pump, the flaps and spoilers will return to the position prior to power shutoff provided their control switch settings have not been changed.



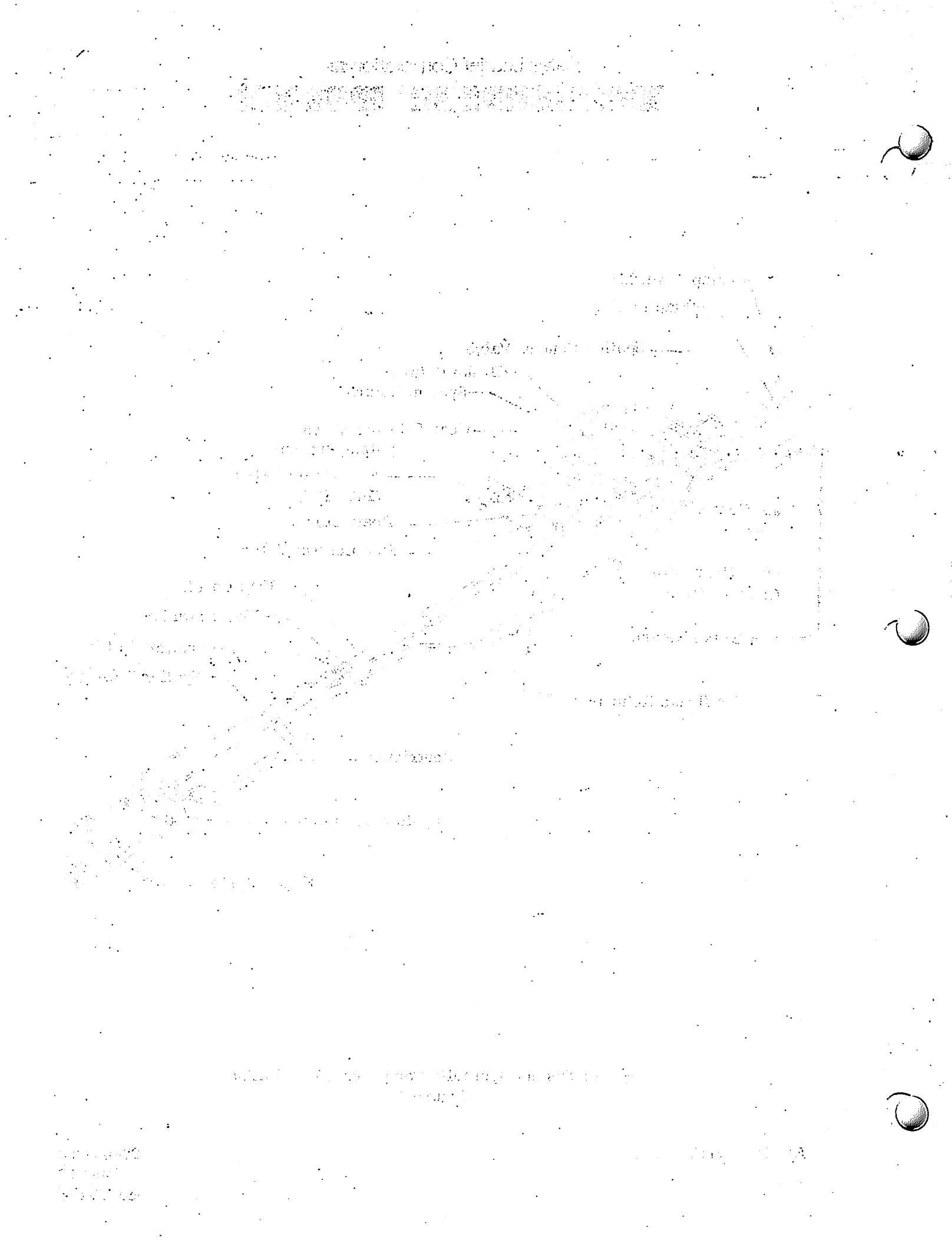
Spoiler System Electrical Control Schematic
Figure 1



Spoiler System Hydraulic System Schematic
Figure 2



Spoiler System Hydraulic Components Installation
Figure 3



SPOILER - MAINTENANCE PRACTICES

1. Removal/Installation

CAUTION: ON AIRCRAFT EQUIPPED WITH RAS, WHEN SPOILER OR SPOILER ACTUATOR IS REPLACED, THE STALL WARNING INSPECTION/CHECK MUST BE PERFORMED. IF THE REQUIREMENTS OF THE INSPECTION/CHECK ARE NOT MET, THEN THE ENTIRE ADJUSTMENT/TEST MUST BE PERFORMED.

NOTE:
- Removal and Installation procedures for left and right spoilers are identical.
- To remove spoilers, flaps and spoilers must be extended.
- Spoiler rigging shall be accomplished after spoiler or spoiler actuator replacement or at any time spoiler actuator rod end adjustment is changed.

A. Remove Spoiler (See figure 201)

WARNING: SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM THE AIRCRAFT WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST AND THEN REMOVE ELECTRICAL POWER.

(1) Extend spoilers and flaps.
(2) Disconnect spoiler bonding jumper(s) from hinge support(s) as shown.
(3) Remove bolts and nuts attaching actuator arm assembly to spoiler control bracket.
(4) Remove inboard and outboard hinge bolts securing spoiler hinges in spoiler hinge supports and remove spoiler from wing.

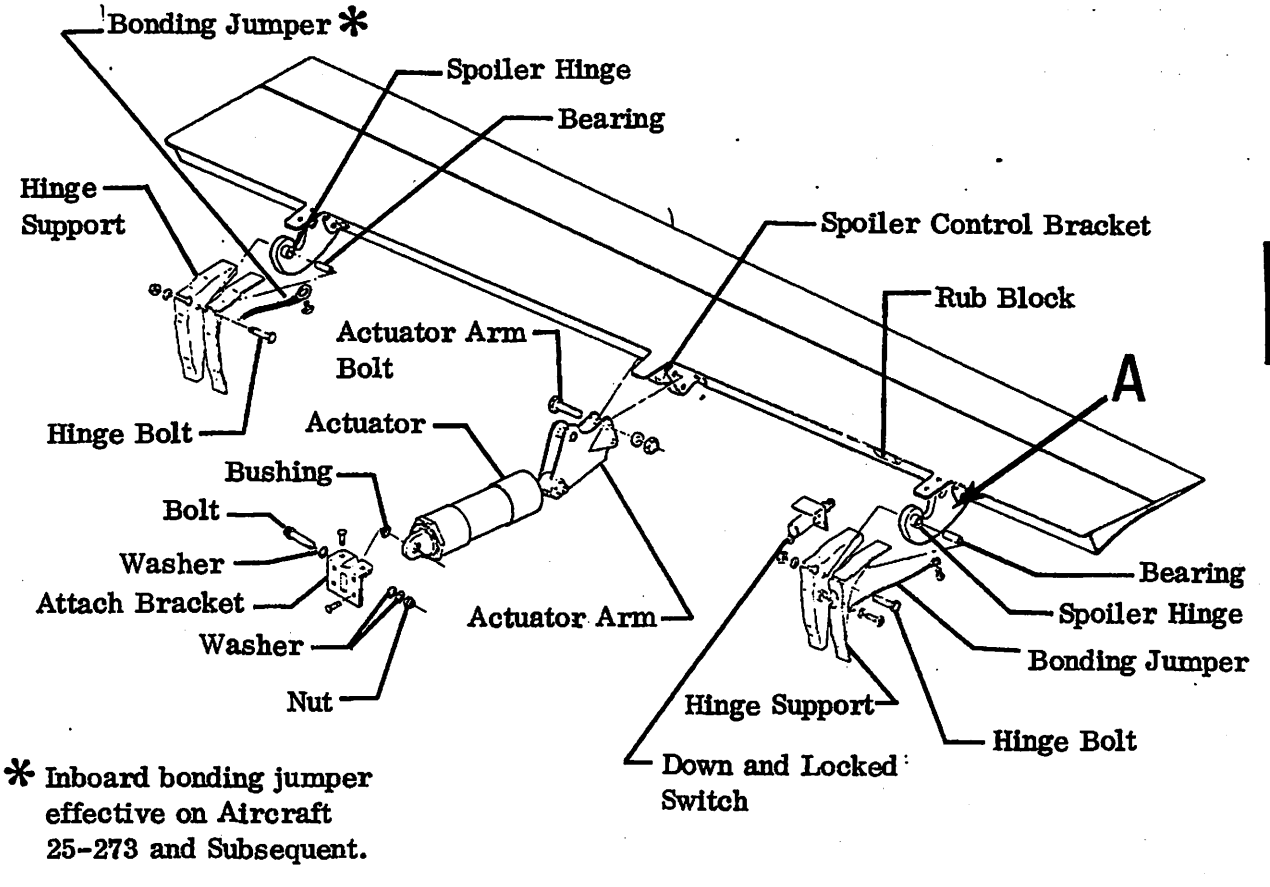
B. Install Spoiler (See figure 201)
(1) With flaps down, align spoiler on wing and insert inboard and outboard spoiler hinges in spoiler hinge supports on wing.
(2) Secure spoiler hinges in hinge supports using attaching hardware.
(3) Attach spoiler control bracket to actuator arm assembly using attaching parts.
(4) Connect spoiler bonding jumper(s) to hinge support(s).

2. Adjustment/Test
A. Rig Spoilers (See figure 201)
(1) Assure that spoiler actuator is secured to wing attachment bracket and that actuator rod end is secured to actuator arm assembly.

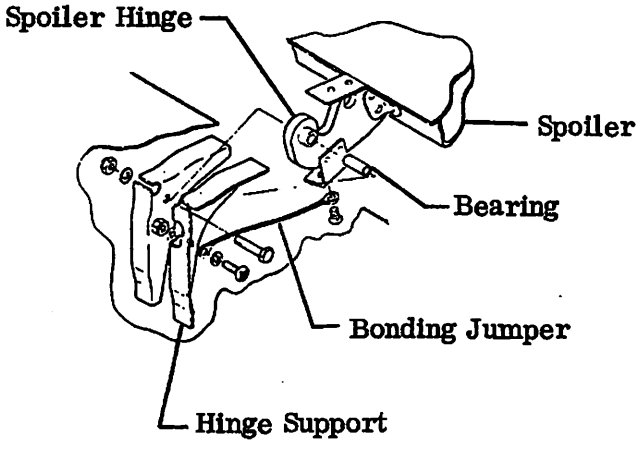
CAUTION: SCREW ACTUATOR ROD END WELL OUT OF ACTUATOR TO AVOID SPOILER OR WING STRUCTURE DAMAGE.

(2) Apply hydraulic pressure to aircraft and extend flaps.
(3) Cycle spoilers. Observe, on retraction, that spoiler fairs to wing surface.
(4) If necessary, loosen jammut on actuator rod end and remove attaching bolt on actuator arm assembly and adjust actuator rod end until spoiler becomes faired with wing.
(5) Tighten jammut on actuator rod end and safety. Attach actuator rod end to actuator arm assembly.
(6) Place protractor on spoiler surface and set protractor on zero.
(7) Cycle spoilers and check for specified travel and freedom of movement. Spoilers must be synchronized within a maximum travel differential of 1°.
(8) With flaps extended and spoilers retracted, remove safety wire and loosen jammuts and safety wire per double twist method.
(9) While contacting spoiler rub block, adjust jammuts until switch actuates and spoiler warning light extinguishes. Adjust jammuts one revolution beyond switch actuation point. Tighten jammuts and safety wire per double twist method.
(10) Extend flaps and spoilers. Check for a minimum of 1/4-inch clearance between aft actuator hydraulic fitting and switch cam on flap sector. If clearance is insufficient, proceed as follows:
 (a) Cut safety wire and loosen actuator head lockmut.
 (b) Rotate actuator until sufficient clearance is obtained.
 (c) Tighten lockmut and safety wire.

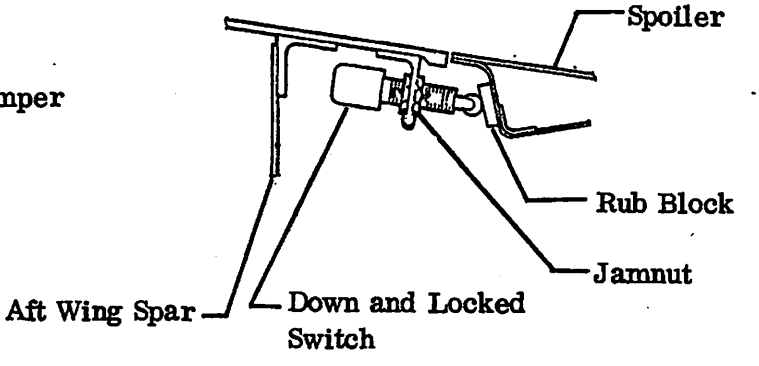
3. Inspection/Check
A. Operational Check of Spoilers
(1) Lower tailcone access door.
(2) Attach a hydraulic power unit to pressure and return quick-disconnects.
(3) Set Battery Switch(es) on and Spoiler Switch to EXT.
(4) Observe that when spoilers extend, the SPOILER warning light illuminates and remains illuminated.
(5) Set Spoiler Switch to RETRACT.
(6) Observe that when spoilers retract, SPOILER warning light extinguishes.
(7) Set Battery Switch(es) to OFF.
(8) Disconnect hydraulic power unit and close tailcone access door.



* Inboard bonding jumper effective on Aircraft 25-273 and Subsequent.



Detail A



Spoiler Installation
Figure 201

SPOILER ACTUATOR - MAINTENANCE PRACTICES

1. Removal/Installation

CAUTION: ON AIRCRAFT EQUIPPED WITH RAS, WHEN SPOILER OR SPOILER ACTUATOR IS REPLACED, THE STALL WARNING INSPECTION/CHECK MUST BE PERFORMED. IF THE REQUIREMENTS OF THE INSPECTION/CHECK ARE NOT MET, THEN THE ENTIRE ADJUSTMENT/TEST MUST BE PERFORMED.

A. Remove Spoiler Actuator (See figure 201)

WARNING: SPOILERS WILL SLAM DOWN WHEN ELECTRICAL POWER IS REMOVED FROM THE AIRCRAFT WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER FIRST AND THEN REMOVE ELECTRICAL.

(1) Extend spoilers and lower flaps for access to spoiler control arm.
(2) Depressurize hydraulic system.
(3) Remove spoiler actuator access panel.
(4) Disconnect and cap extend and retract hydraulic lines.
(5) Remove the bolt, nut, and washer attaching actuator rod end to the control arm. Remove attaching parts that attach forward end of actuator to wing structure and remove actuator from aircraft.

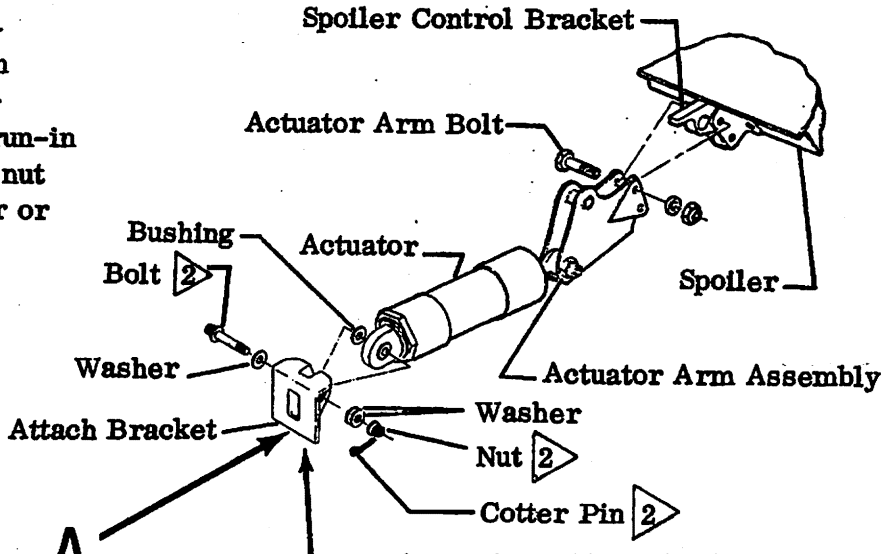
B. Install Spoiler Actuator (See figure 201)
(1) Position actuator in wing and secure forward end of actuator to wing structure.
(2) Connect actuator rod end to control arm.
(3) Connect retract and extend hydraulic lines to actuator ports.
(4) Rig spoilers.
(5) Install access covers.

2. Approved Repairs

NOTE:
- Approved repairs of the spoiler actuator consist of replacing O-rings, backup rings, and felt wiper. These new items must be immersed in MIL-H-5606 hydraulic fluid immediately prior to installation.
- Prior to assembly, inspect to assure that all O-ring grooves are free of gouges, scratches, and burrs. Assure that all items are thoroughly clean prior to installation.

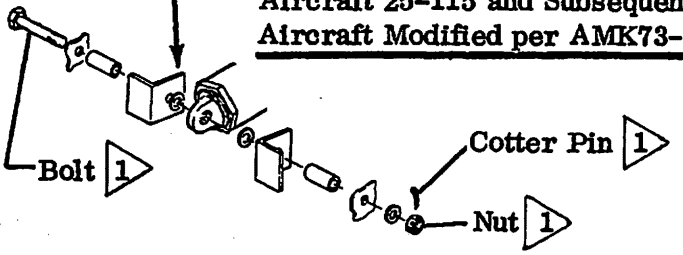
A. Disassemble Spoiler Actuator (See figure 202)
(1) Cut safety wire and loosen jammut sufficiently to release rod end lock. Remove rod end from piston rod.

2 Effective 25-346 and Subsequent, install cotter pin and torque nut to 30 inch-pounds minimum above run-in torque. Replace bolt and nut if inspection reveals wear or abrasion.

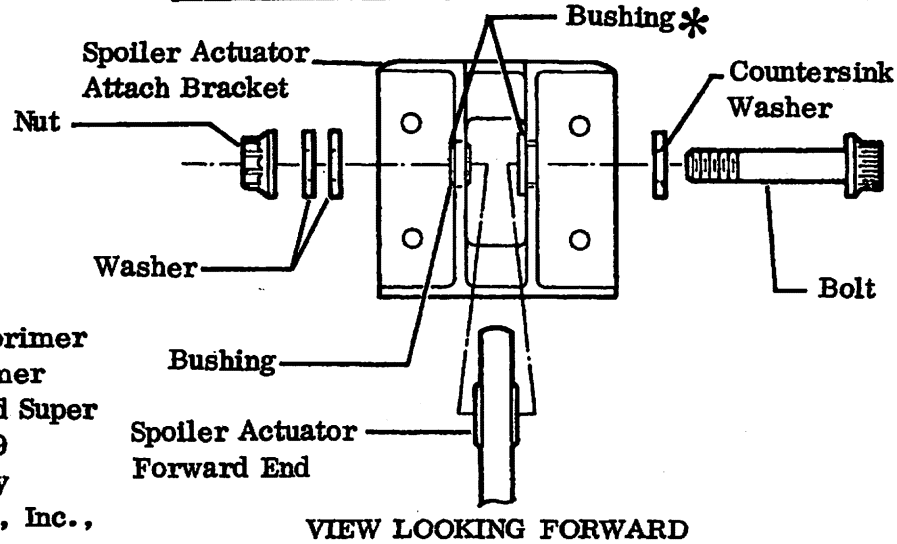


Aircraft 25-115 and Subsequent and Aircraft Modified per AMK73-1.

1 On Aircraft 25-061, 25-070 thru 25-114 Not Modified per AMK73-1, replace bolt, nut, and cotter pin in accordance with current replacement intervals specified in Chapter 5.



Aircraft 25-061, 25-070 thru 25-114 Not Modified per AMK73-1.



VIEW LOOKING FORWARD
(Typical for Both Wings)

* Install bushings with wet primer Super Koropon Green Primer #515-005 Color #34151 and Super Koropon Curing Solution #9 #910-014 (Manufactured by Desoto Chemical Coatings, Inc., Chicago, Illinois.)

Detail A

Spoiler Actuator Installation
Figure 201

4-32D-1
A8-75A

(2) Mark clocked position of head end on barrel assembly and number of threads protruding from barrel assembly before removing head end. Loosen locknut on head end sufficiently to allow head end to be screwed out of barrel assembly.

(3) Slide piston and locking balls out of barrel assembly. The locking balls will fall free as piston is removed from barrel assembly.

(4) Remove O-rings and backup rings from piston rod.

WARNING: THE PLUNGER IS SPRING-LOADED IN THE HEAD END. EXTREME CARE MUST BE TAKEN WHEN REMOVING PLUNGER FROM THE HEAD END TO PREVENT EITHER PART FROM SPRINGING OUT OF CONTROL. OTHERWISE BODILY INJURY MAY RESULT.

(5) Remove retaining ring, plunger, and spring from head end.

(6) Remove O-rings and backup rings from plunger.

(7) Remove felt wiper, O-ring, and backup ring from barrel assembly.

(8) Clean all parts with Stoddard solvent, or equivalent. Dry parts with clean, dry compressed air.

(9) Inspect barrel assembly, piston rod, plunger, and head end for nicks, scratches, or excessive wear. Buff out minor defects with an oil stone, then polish with crocus cloth.

B. Assemble Spoiler Actuator (See figure 202)

(1) Install backup ring, O-ring, and felt wiper in barrel assembly.

(2) Install O-rings and backup rings on plunger. Insert spring into plunger cavity.

(3) Insert plunger and spring into head end and secure with retainer ring.

(4) Install O-rings and backup rings on the piston rod.

(5) Coat outside and inside diameters of piston rod with AN-P-51 (vaseline) as required to hold locking balls in place in the holes in the piston rod. Install balls in piston rod.

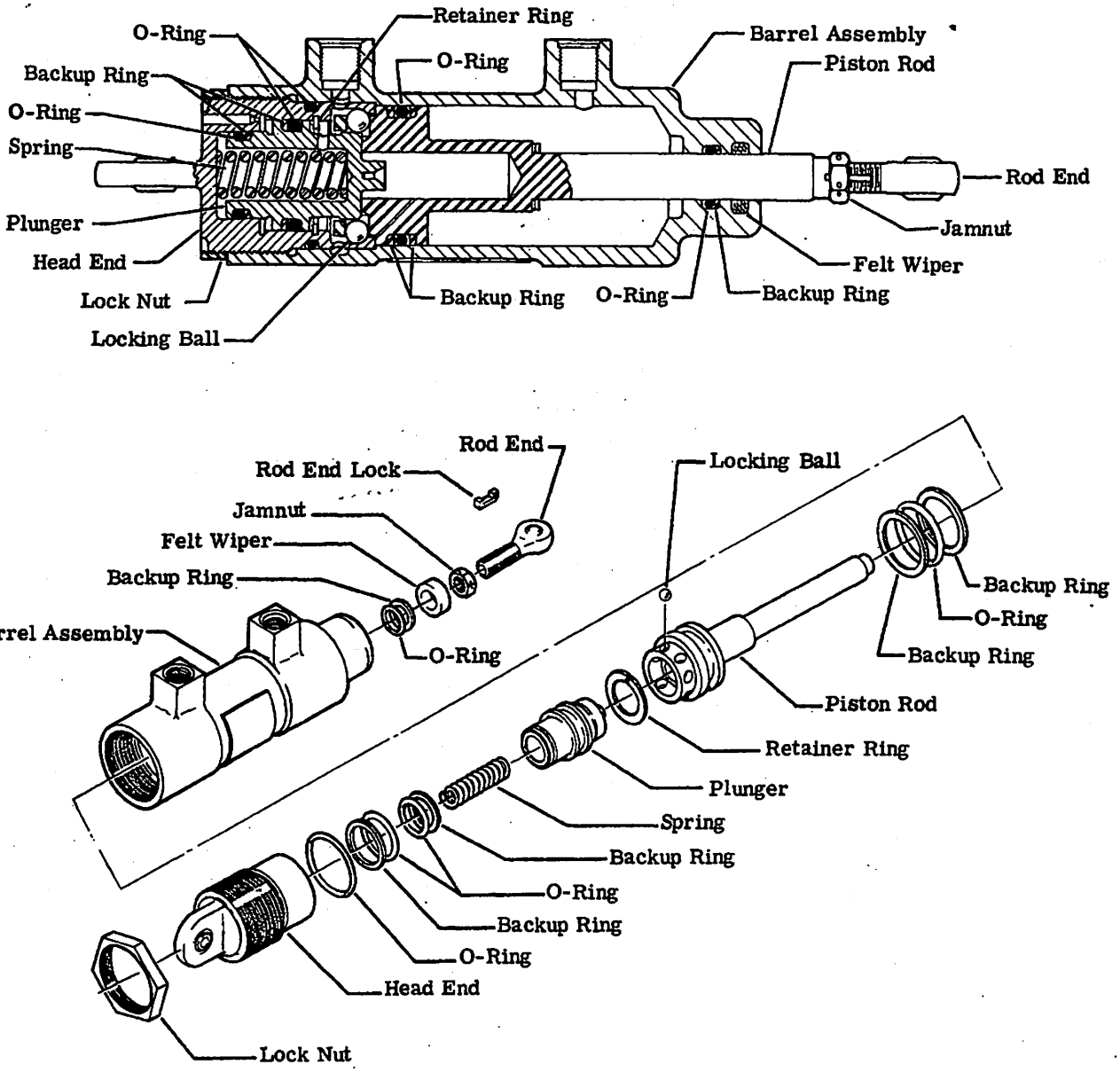
(6) Carefully insert piston rod into barrel assembly assuring locking balls are not dislodged.

CAUTION: WHEN INSTALLING HEAD END INTO BARREL ASSEMBLY ASSURE THAT THE HYDRAULIC PORTS ARE CLOCKED 90° IN RELATION TO THE CLEVIS PORTION OF THE HEAD END.

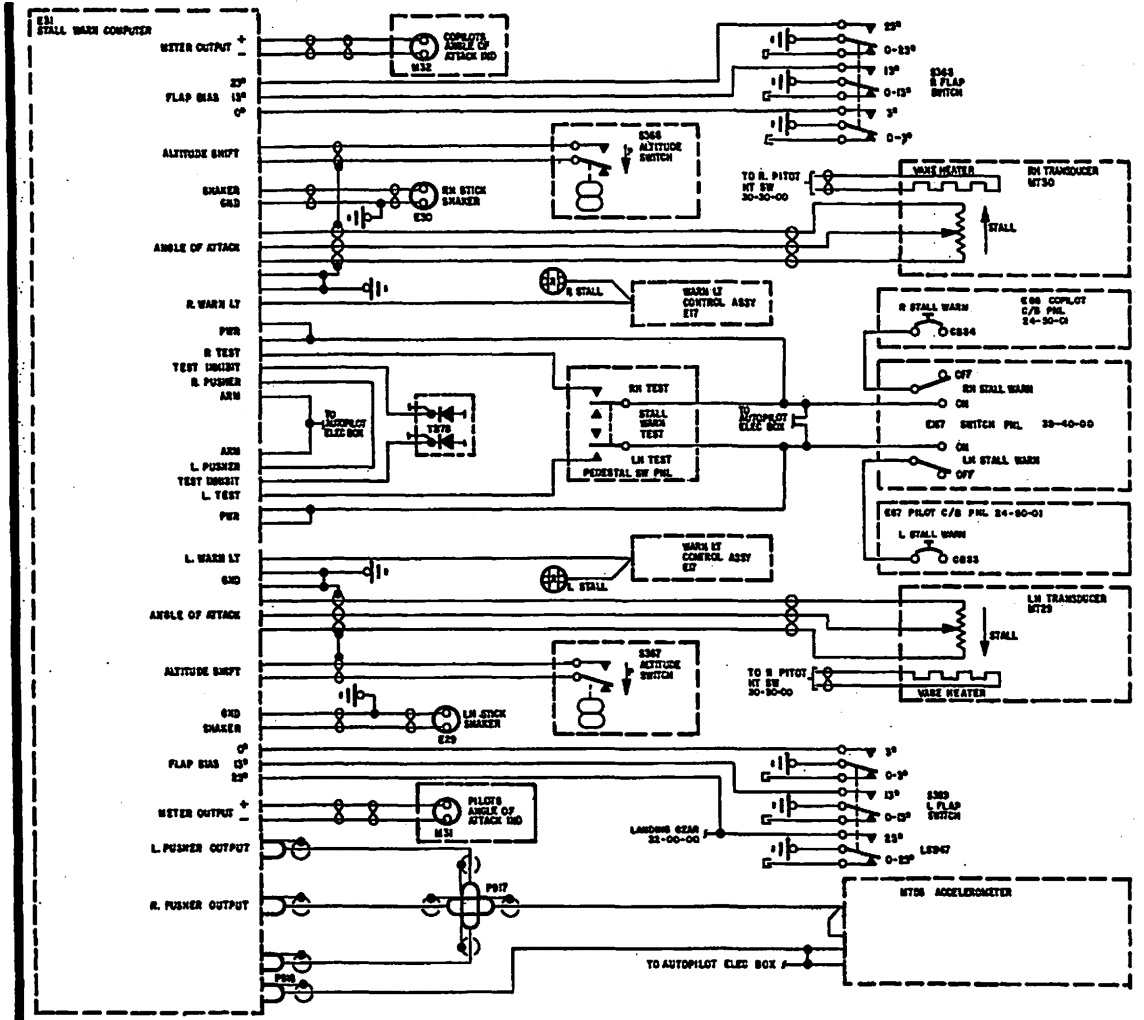
(7) Screw head end into barrel assembly leaving the same number of threads exposed and so that clevis end is clocked as prior to removal. Install locknut.

(8) Install rod end, rod end lock, and jammut on piston rod. Safety wire rod end lock.

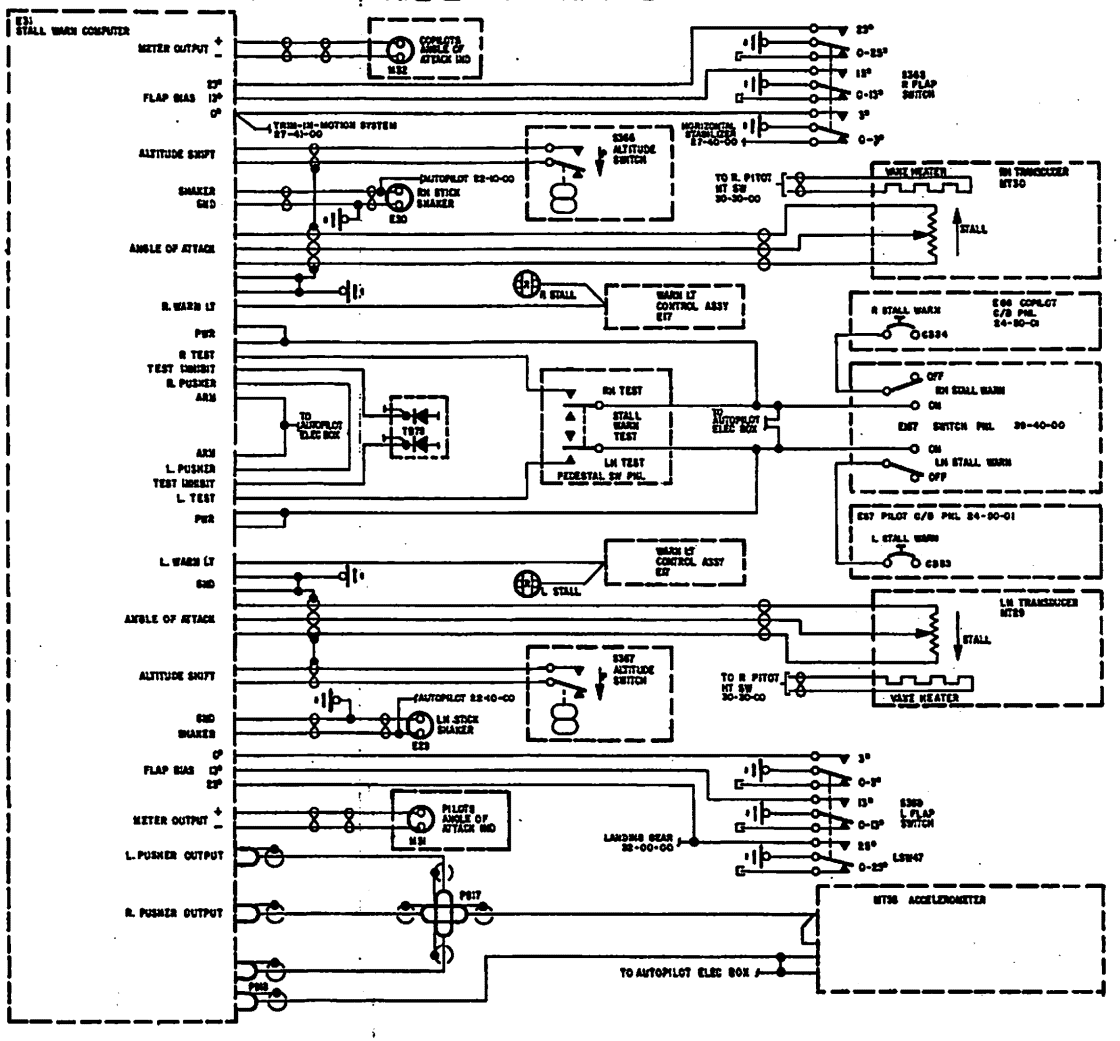
(9) Connect actuator to hydraulic test bench and cycle several times at 2250 psi (maximum). Maximum allowable leakage is one drop.



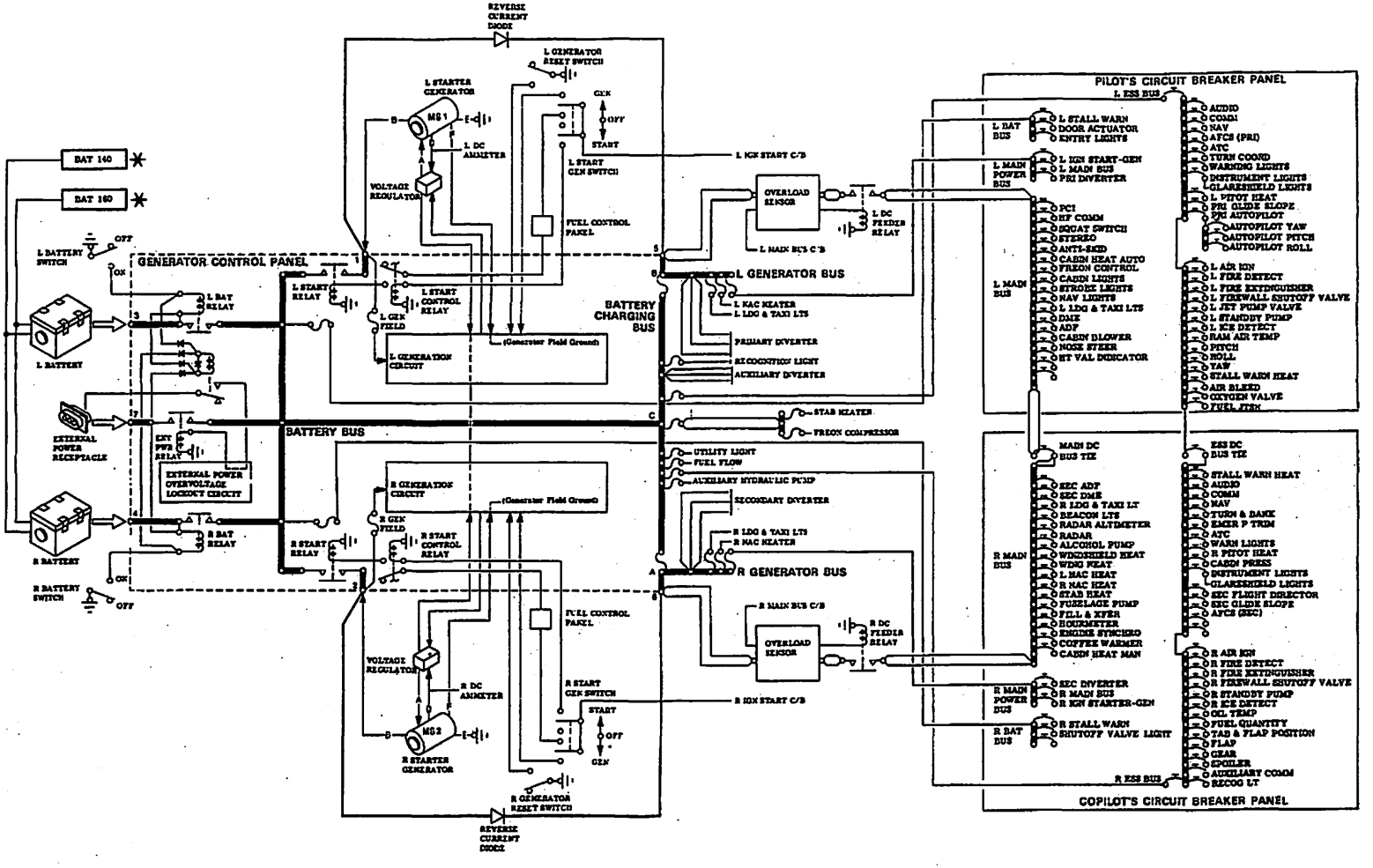
Spoiler Actuator Assembly
Figure 202



Stall Warning Electrical Control Schematic
Figure 4

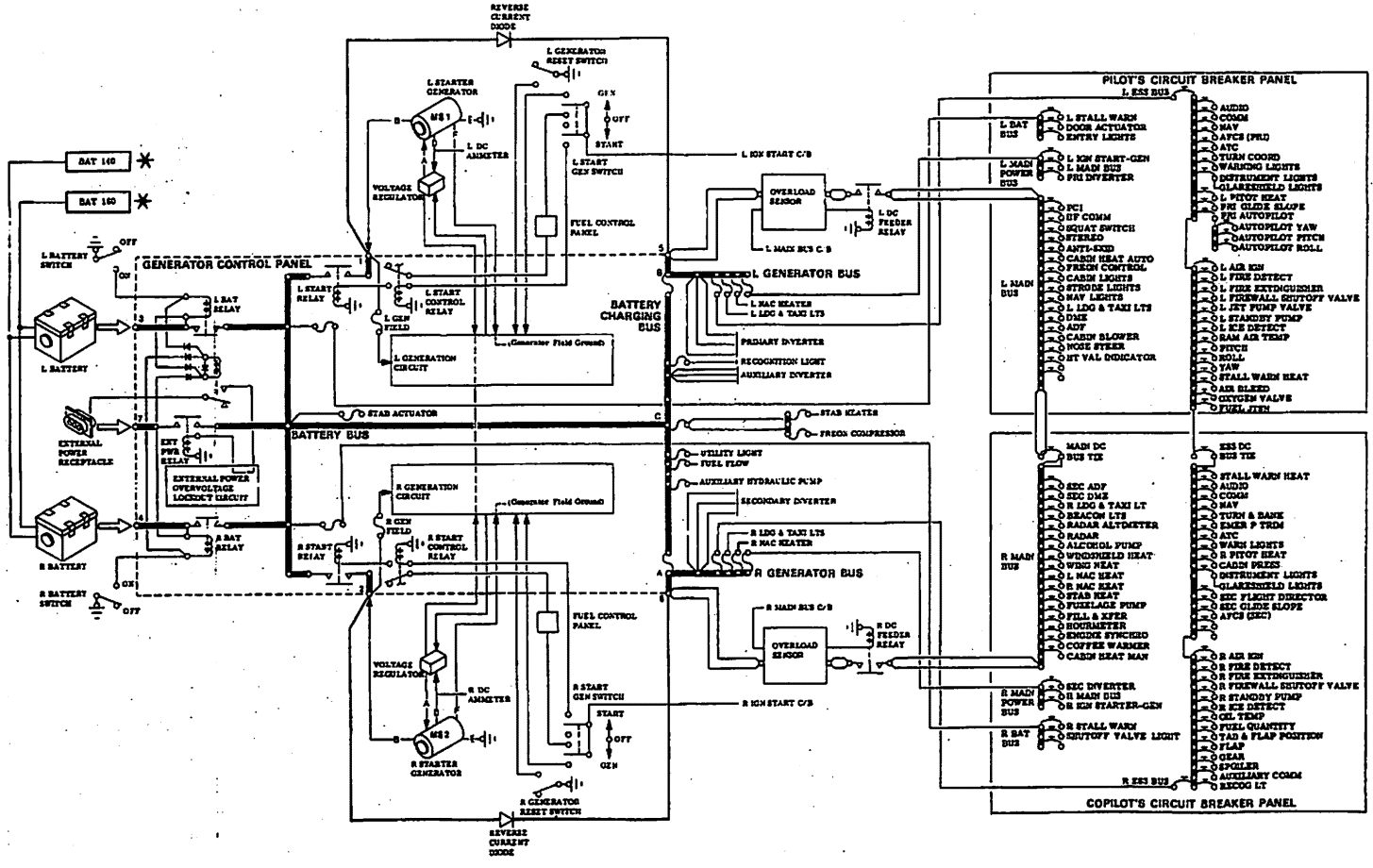


Stall Warning Electrical Control Schematic
Figure 5



DC Generation Electrical Control Schematic
Figure 1 (Sheet 2 of 3)

* Battery overheat warning lights are effective only on aircraft equipped with nickel-cadmium batteries.

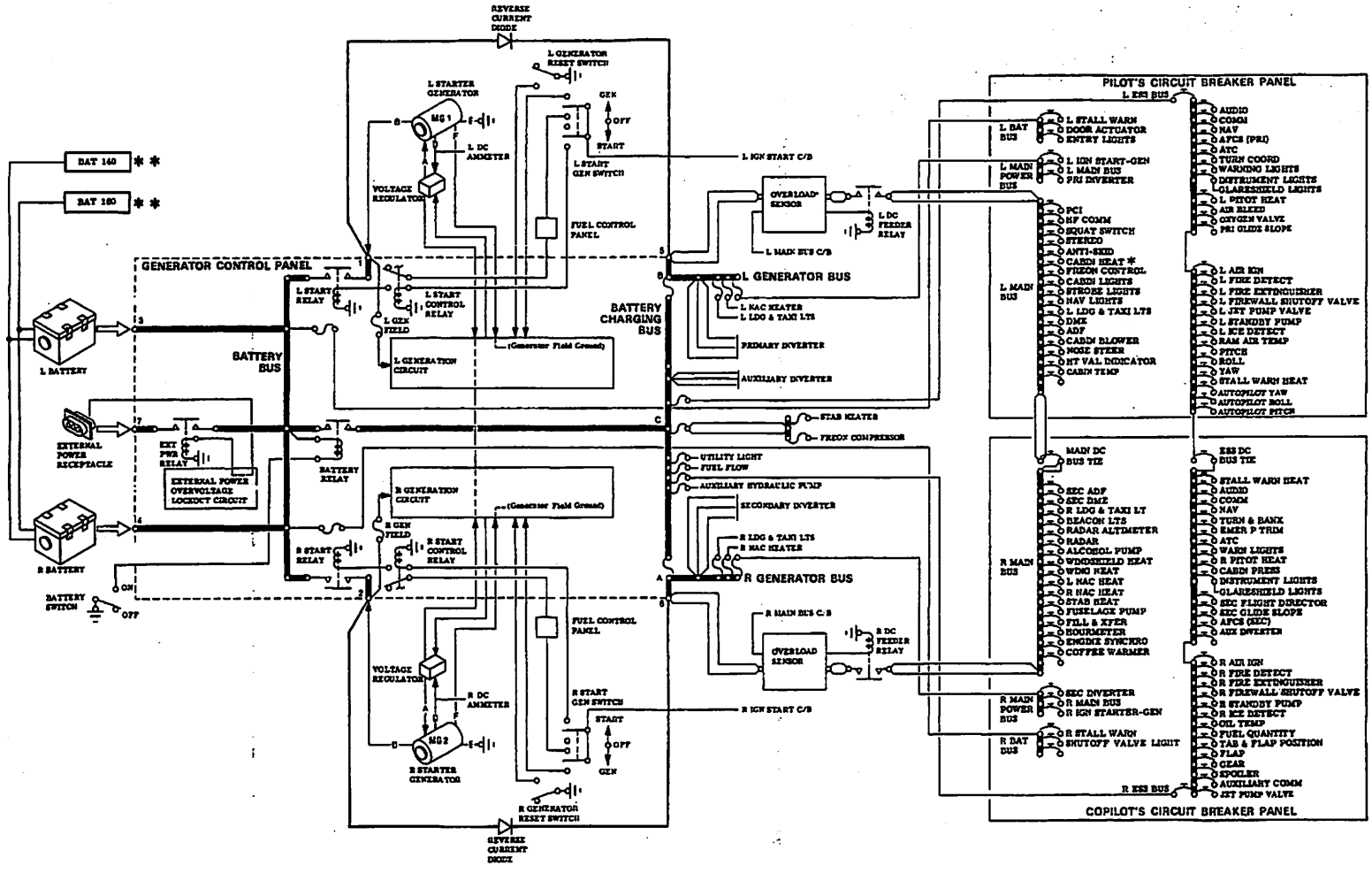


DC Generation Electrical Control Schematic
Figure 1 (Sheet 3 of 3)

* Battery overheat warning lights are effective only on aircraft equipped with nickel-cadmium batteries.

B. Generator Caution Light Operation (See figure 2.)

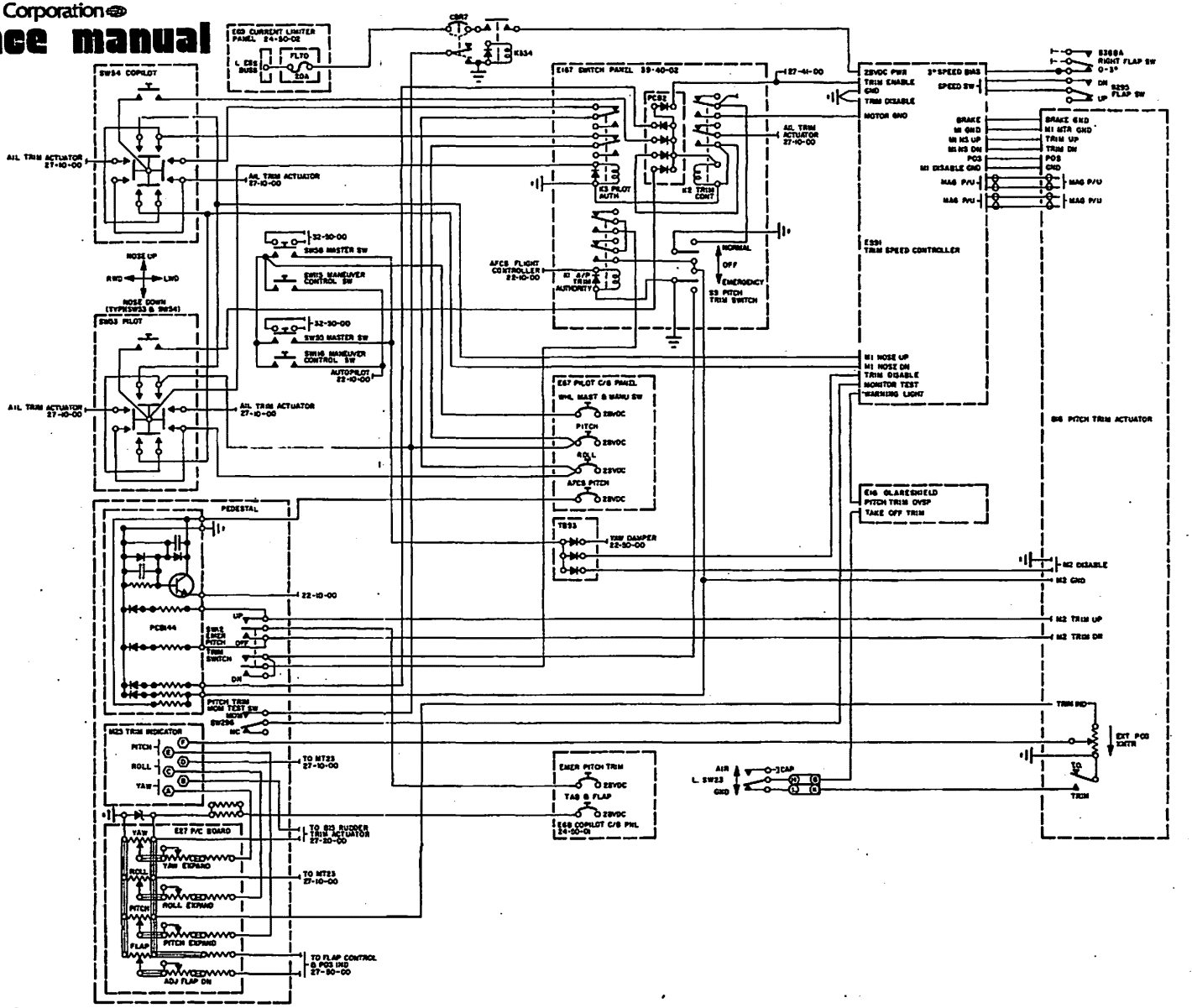
(1) The generator caution lights (LH) are lighted when a ground is applied to the lights through a set of contacts of deenergized relay (K7). When the Starter-Generator Switch is set to GEN, relay (K5) is energized and applies generator output voltage directly to relay (K7) and opens the generator ground circuit. Operation of the RH generator caution light is identical to the LH except relays (K6 and K8) are used.



DC Generation Electrical Control Schematic
Figure 1 (Sheet 1 of 3)

* On Aircraft 25-084 thru 25-089 and 25-091 thru 25-102, separate circuit breakers are provided for manual and automatic cabin heat.

** Battery overheat warning lights are effective only on aircraft equipped with nickel-cadmium batteries.



Pitch Trim Electrical Control Schematic
Figure 2 (Sheet 3 of 3)

EFFECTIVITY: Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement

(either nose up or nose down) is applied through the Emergency Pitch Trim Switch to the secondary pitch trim motor.

B. Aircraft 25-337, 25-342 and Subsequent and prior Aircraft modified for Horizontal Stabilizer Trim and Autopilot Improvement.

(1) Normal Operation

(a) With the Pitch Trim Switch set to NORM, a standby ground circuit is completed through the Pitch Trim Switch to the de-energized contacts of the trim control relay. Twenty-eight vdc is applied from the PITCH circuit breaker to the Pilot's Trim and Trim Arming Switch. When the Pilot's Trim and Trim Arming Switch is depressed and set to either NOSE UP or NOSE DN, 28 vdc is applied to the pilot's authority relay, the trim control relay, and the pitch trim actuator primary (M1) motor through the trim speed controller. The pilot's authority relay energizes to remove power from the copilot's Trim and Trim Arming Switch while the trim control relay energizes to complete a circuit to the pitch trim actuator primary motor through the pitch trim speed controller and then to ground.

(b) The low speed of the primary motor is achieved by reducing current through the trim speed controller when the flaps are up. Should a failure occur which would allow full-speed motor operation with the flaps up, a self-monitoring circuit will illuminate the Pitch Trim Overspeed warning light on the glareshield.

(c) At take-off, with the squat switches in ground mode, if the horizontal stabilizer pitch angle is beyond prescribed take-off up or down limits, the applicable up or down limit switch in the stabilizer actuator linear potentiometer will be closed, making a circuit to ground through the left squat switch to the TAKE-OFF TRIM warning light.

(d) With the flaps up and the Pitch Trim Switch set to NORM, 75 (+3, -0) seconds are required for the actuator to move from stop to stop. With the flaps down and the Pitch Trim Switch set to NORM, 16 (+4, -0) seconds are required for the actuator to move from stop to stop.

(e) The actuator incorporates a linear potentiometer which follows actuator motion. The potentiometer supplies a signal to the pitch trim position indicator and modulates a circuit to the trim-in-motion detector. (Refer to 27-41-00.)

(f) During normal operation, the speed controller and pitch trim actuator primary motor are energized through the closed contacts of the pitch trim actuator overload sensor relay and the PITCH circuit breaker. If an overload condition exists, the thermal breaker contained within the sensor will position the switch contacts to remove power from the relay and ground the PITCH circuit breaker. The overload sensor will reset when the thermal breaker has cooled. However, the 28 vdc required to energize the